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# AH-1G DESIGN AND OPERATIONAL FLIGHT LOADS STUDY

Max E. Glass, et al

Bell Helicopter Company

Prepared for:

Army Air Mobility Research and Development Laboratory

January 1974

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13. ABSTRACT

This report compares AH-1G helicopter Southeast Asian mission profiles with the original engineering frequency-of-occurrence spectrum and the Navy AR-56 spectrum for attack helicopters. Fatigue lives calculated using the Southeast Asian profile are compared with those determined using the original frequency-of-occurrence spectrum. The development cycle of the Bell Helicopter Company Model 540 rotor system is reviewed, and the fatigue design methods used are presented. Maximum one-time occurrences measured in the Southeast Asian operational survey are compared with those specified in the AH-1G structural design criteria and those measured in structural demonstration flight tests. Recommendations are made regarding future mission surveys, the structural design criteria for attack helicopters, and the upgrading of rotor loads prediction capability.



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The information presented herein is the result of an analytical effort to derive improved structural design criteria for gunship-type helicopters based upon flight parameters measured on gunship helicopters operating in Southeast Asia. This is one of four similar efforts being conducted concurrently to develop improved criteria for observation, crane, and transport as well as gunship-type helicopters.

The report has been reviewed by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound. It is published for the exchange of information and the stimulation of future research.

This program was conducted under the technical management of Mr. Herman I. MacDonald, Jr., Technology Applications Division.

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# AH-1G DESIGN AND OPERATIONAL FLIGHT LOADS STUDY

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## FOREWORD

This report, "AH-1G Design and Operational Flight Loads Study," was prepared by Bell Helicopter Company, Fort Worth, Texas, for the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under Contract DAAJ02-72-C-0099, Task 1F162204A17002. Mr. Herman MacDonald was the contract technical representative.

The authors acknowledge the help of Bell Helicopter Company's Mr. M. J. McGuigan, Chief of Structural Design, for his contributions to the study. For their contributions in the discussion of limiting factors and maximum one-time occurrences, the authors are grateful to Bell Helicopter Company personnel:

Mr. G. Colvin, Senior Test Pilot
Mr. D. Bloom, Senior Test Pilot
Mr. J. Duhon, Group Engineer, Aerodynamics
Mr. C. Livingston, Group Engineer, Stability and Control

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# INTRODUCTION

An essential factor in determining fatigue life is an estimate of the frequency of occurrence of the various flight conditions the helicopter will experience in service. In the preparation of the original engineering development frequency-of-occurrence spectrum for the AH-1G helicopter, as used in Reference 1, full advantage was taken of all information available on this subject. This included considerable past experience in the preparation of similar spectrums for other helicopters, both military and civilian. Data were also available from statistical surveys conducted on helicopters in actual service. Bell Helicopter, under Army funding, has conducted several of these statistical surveys on the YH-40 (HU-1A), HU-1B, and YUH-1D helicopters under normal operating conditions which are reported in References 2, 3, and 4. All of these surveys include VGH (velocity, acceleration, and altitude) data as well as important channels of strain gage data for oscillatory load information. Some other useful statistical publications were available for both civilian and Army helicopters, including the UH-1B, HU-1A, and H-13H as presented in References 5, 6, and 7, the H-13, and UH-1A. A suggested frequency-of-occurrence spectrum for helicopters is also included in Civil Aeronautics Manual 6, "Rotorcraft Airworthiness," Appendix A, "Service Life Determination."

In addition to the background of statistical data and previous experience, discussions with Army helicopter pilots with combat experience and with AH-1G test pilots were considered in evaluating the probable operations of this helicopter. All of the above information was used in deriving the original engineering development frequency-of-occurrence spectrum for the AH-1G shown in Table I. A comparison with the Army logistical evaluation test plan for this helicopter showed good general agreement in proportion of time allotted to cruise and maneuvers, although the fatigue life determination spectrum was weighted more heavily toward the high airspeed capabilities of the helicopter.

The AH-1G helicopter, Figure 1, has undergone heavy usage in the combat zones of Southeast Asia. In an effort to gain information on the operational environment of the AH-1G in combat situations, the flight loads investigation outlined in Reference 8 was conducted. These data present an opportunity to investigate the relationship between the original predicted frequency-of-occurrence spectrum and the actual operational environment of the helicopter. This task is addressed in subsequent sections of this report, with the ultimate purpose being to calculate fatigue lives for various components to study the impact of spectrum variations and modifications.

Figure 1. Bell Model AH-1G Helicopter.

To provide some insight into the fatigue design problem, the development of the 540 rotor system is reviewed and the approach used in sizing the components of this system for fatigue loading is discussed. The fatigue design methods are then evaluated in light of operational experience to determine if weaknesses in the method could be related to a shorter than predicted fatigue life encountered for a major component. The purpose of this section is to show the major variables, component fatigue strength, oscillatory loads, and frequency-of-occurrence spectrum in a total perspective. In this manner, the significance of each variable can be examined and its importance at various stages of the design-development cycle can be evaluated.

In addition to the frequency-of-occurrence spectrum comparison, a study of maximum one-time occurrences was also conducted. The maximum one-time occurrences of the load-significant parameters measured in the mission profile survey are compared with the values specified in the structural design criteria, the aircraft operating limitations, and those measured in the helicopter flight structural demonstration. A discussion of what factors probably caused each maximum one-time value is given, and a method for predicting these limiting parameters in future attack-type helicopters is suggested.

## ANALYSIS OF DATA

The operational data presented in Reference 8 were accumulated in two data samples. The first sample represented 201.8 flight hours, and the second sample represented 206.5 flight hours. In order to perform a comparative analysis of these data with the original frequency-of-occurrence spectrum, the mission data were converted to a 100-hour base. This conversion was accomplished by using a weighted percentage for each data increment of each parameter. Having accomplished this conversion, the combat operational data were then compared directly with the data from the original frequency-of-occurrence spectrum.

### MISSION SEGMENTS

The operating environment of the AH-1G has been divided into four mission segments for the purpose of comparing original spectrum data with operational data. These segments are: (1) ascent, (2) maneuver, (3) descent, and (4) steady state. The ascent stage is defined as that portion of flight from takeoff to cruise altitude and any ascents to other altitudes. The maneuvering segment includes all attitude and direction changes, power transitions, and any conditions in which weapon firing occurred. The descent stage includes the time in descent, flare, and landing. Steady state includes all stabilized flight such as cruise, hover, and stabilized autorotation.

The data presented in Figure 2 show a significant difference between the maneuver and steady-state time distributions of the original frequency-of-occurrence spectrum and the combat operational data. Some of this difference can possibly be attributed to assigning a few flight conditions from the original spectrum into different segments than was done in Reference 8. However, most of the difference is thought to be the result of a change in helicopter mission description. In preparing the original frequency-of-occurrence spectrum, it was anticipated that the ship would take off, cruise to the target area, accomplish the mission task, and cruise back to its base. The combat operational data indicate frequent excursions during the outgoing and return cruise operation. These excursions were apparently to search for additional activity outside the prescribed mission.

### GROSS WEIGHT

The gross weight envelope for the AH-1G helicopter, set forth by Bell Helicopter Company, was from 6500 pounds to a maximum of 9500 pounds. In establishing the fatigue lives for the AH-1G components, the following gross weight distribution was

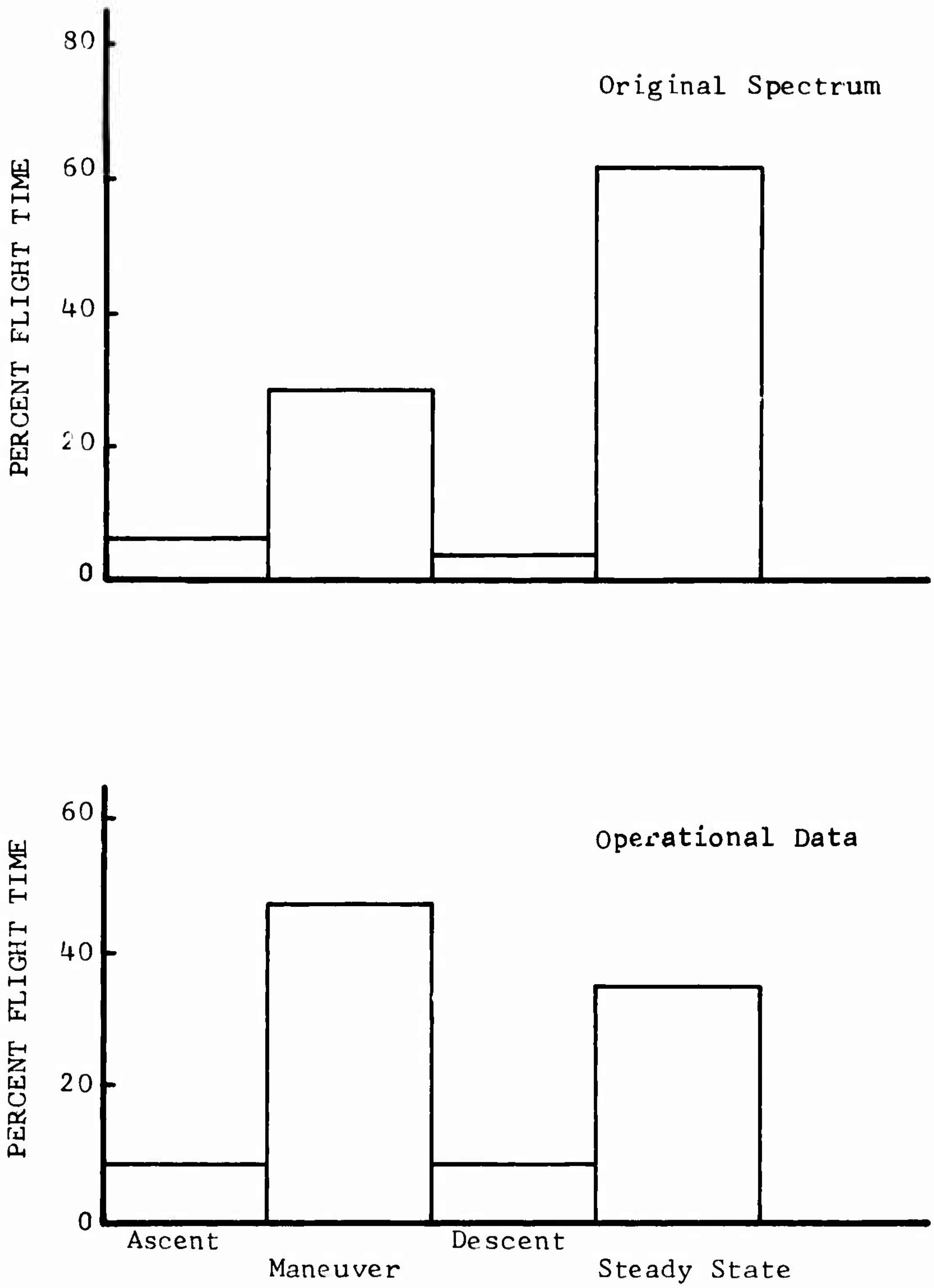
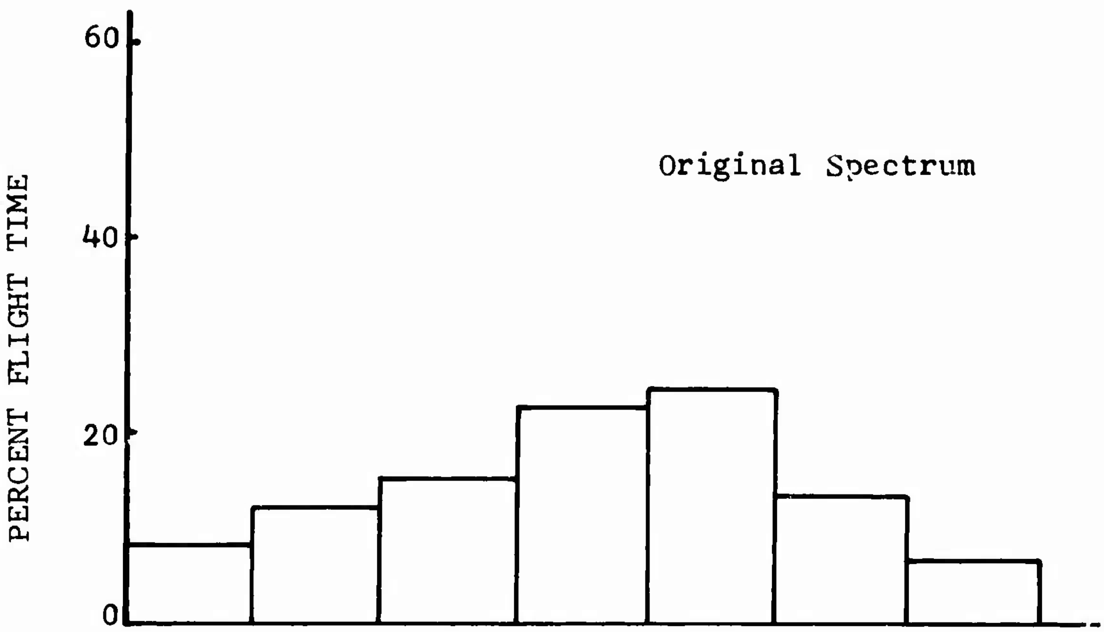



Figure 2. Distribution of Flight Time in Mission Segments for the Original AH-1G Spectrum and the Operational Data.



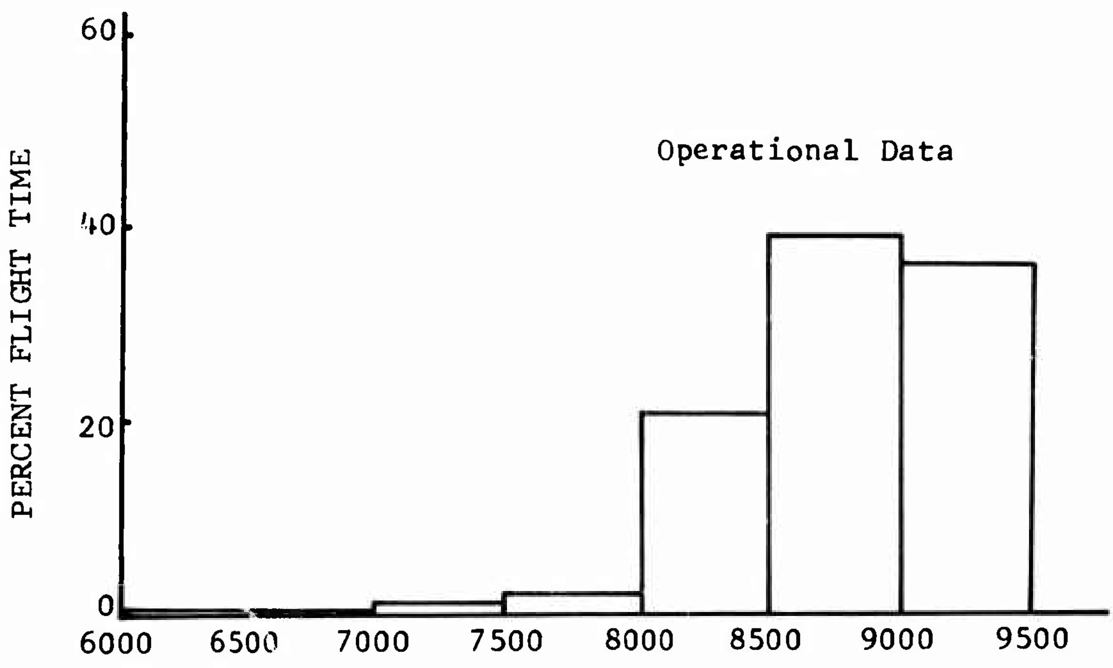




Figure 3. Distribution of Flight Time for Gross Weights for the Original AH-1G Spectrum and the Operational Data.

used: light gross weights to 7000 pounds, 20 percent; medium gross weights from 7001 to 8200 pounds, 50 percent; and heavy gross weights from 8201 to 9500 pounds, 30 percent of total flight time. As shown in Figure 3, the gross weight distribution determined from the operational data indicates that considerably more time is spent in the heavy gross weight conditions.

## AIRSPEED

The airspeed distributions presented in Reference 8 were combined into a total combat operational profile shown in Figure 4. These data differ significantly from the airspeed profile for the original spectrum also shown in Figure 4. The combat operational data are distributed about the 90- to 100-knot increment, while the original spectrum data are distributed about the 130- to 140-knot increment. However, the maximum airspeed demonstrated within the original data was 184 knots compared to the combat operations maximum of 186 knots. This indicates that the airspeed range for the original data was sufficient to cover the operational airspeeds of the helicopter, but the airspeed distribution was not accurately predicted.

## OTHER PARAMETERS

Operational data for four other parameters - (1) altitude, (2) rotor speed, (3) torque pressure, and (4) vertical load factor - were available for comparison. The original design spectrum did not include distributions of these parameters; therefore, no comparison is shown.

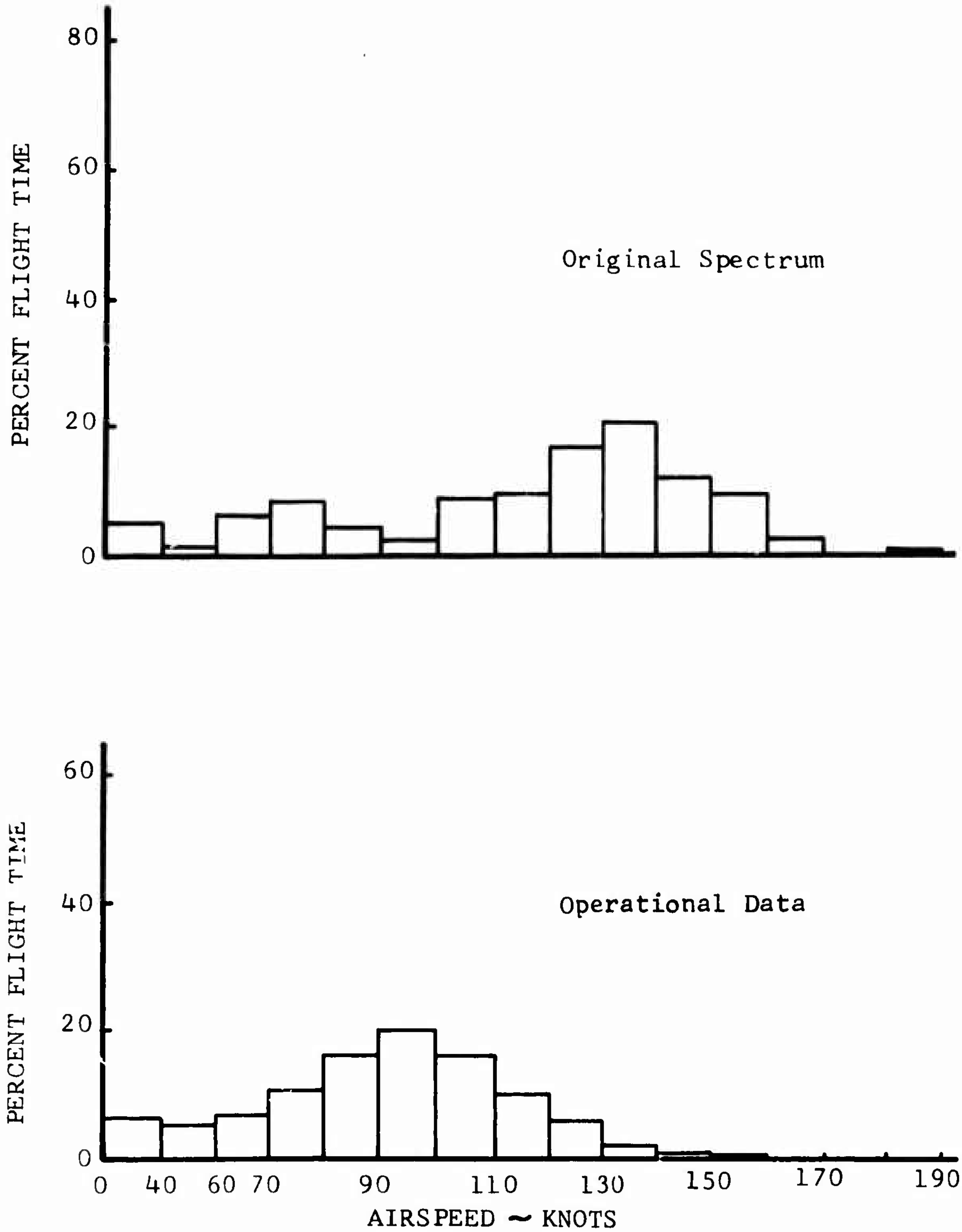



Figure 4. Comparison of Airspeed Distribution for the Original AH-1G Spectrum and the Operational Data.

## PROBLEMS ENCOUNTERED DURING ANALYSIS

Since the original design spectrum did not include distributions of all parameters, an effort was made to compare the operational data to the developmental load level survey data of Reference 9. Several problems arose in comparing those data since the load level survey data were obtained at discrete representative values of some parameters rather than at several values distributed to represent the predicted use of the aircraft. For example, the operational data increments for altitude in Reference 8 were:

- 0 to 1000 feet
- 1001 to 2000 feet
- 2001 to 5000 feet
- 5001 to 10,000 feet
- 10,001 to 15,000 feet

This format may have been best for the mission data. However, altitudes in Reference 9 were condition-entry altitudes only: IGE, 1500, 5000, and 10,000 feet, with no continuous readings. Therefore, comparison was virtually impossible and undoubtedly inconclusive.

A similar problem arose when considering vertical load factor. The mission data were continuously reduced, with all excursions outside the 0.8 - 1.2 g range measured and recorded. However, the original spectrum load factor data were reduced only at the point of maximum g's and therefore do not represent a total data profile. For example, consider a diving maneuver. At the pushover, the load factor drops below 1.0 g. During the dive, the g level returns to the 1.0 g level, and when control inputs are made to pull out, a buildup in excess of 1.0 g will occur. Although the load factor has experienced a range of values from less than 1.0 g to a maximum greater than 1.0 g, the original spectrum data would list only the maximum value. The omission of the remaining data prohibited presentation of a load factor distribution of the original spectrum data.

Another problem area was found in the presentation of the rotor speed data. The mission data increments of Reference 8 are listed below.

- Up to 295 rpm
- 296 to 310 rpm

- 311 to 325 rpm
- 326 to 330 rpm
- 331 to 340 rpm
- 341 to 355 rpm
- 356 rpm and above

This arrangement prevented a detailed comparison since the normal operating range of rotor speed for the AH-1G is from 314 to 324 rpm. Due to the unfortunate choice of increments, all of the normal operation time falls into the third increment, leaving less than 5 percent of the total time to be distributed among the remaining increments.

# OPERATIONAL SPECTRUM DERIVATION

Having analyzed the available data, the original spectrum was used as a baseline and was modified based on the operational data to derive an operational spectrum. The first step in modification was to redistribute the flight time to agree with the mission segment data from Reference 8. When this was accomplished, the airspeed distribution was then investigated. Due to interaction between the mission segments and airspeed, it was not possible to bring the modified spectrum into exact agreement with the operational data. However, the difference is considered insignificant. Table I shows the original engineering frequency-of-occurrence spectrum, Reference 1, and the derived operational frequency-of-occurrence spectrum to be used for reevaluating fatigue lives of selected AH-1G components.

Figures 5 through 9 contain histograms of five data parameters for the purpose of comparing the profiles of the operational data and data from the derived operational frequency-of-occurrence spectrum. Figure 5 shows the new distribution of flight time in the various mission segments. There are slight differences between the operational data of Reference 8 and the derived operational spectrum. These differences could easily be attributed to variations in the flight conditions as assigned to the various mission segments. Therefore, the extent of agreement is considered satisfactory.

Airspeed distributions are presented in Figure 6. Both the derived operational frequency-of-occurrence spectrum and the operational data are distributed about the 90- to 100-knot increment. Also, the relative distributions of both data sets are quite comparable. There is a very small amount of time remaining in the modified spectrum distribution for airspeeds between 180 and 190 knots. This is necessary since the maximum airspeed for the helicopter is in this range and cannot be completely ignored.

Engine delta torque distributions are presented in Figure 7. The only significant difference in these distributions is the larger amount of time in the derived operational spectrum for the 0- to 10-psi increment. This difference is due to the time allotted for autorotation flight and transitions in the derived operational frequency-of-occurrence spectrum. Autorotation is normally an emergency procedure; therefore, little time is spent in this flight mode during normal combat operations. However, the amount of time allocated to autorotation flight in the derived operational spectrum seems necessary to account for all operations including training and emergency procedures practice.

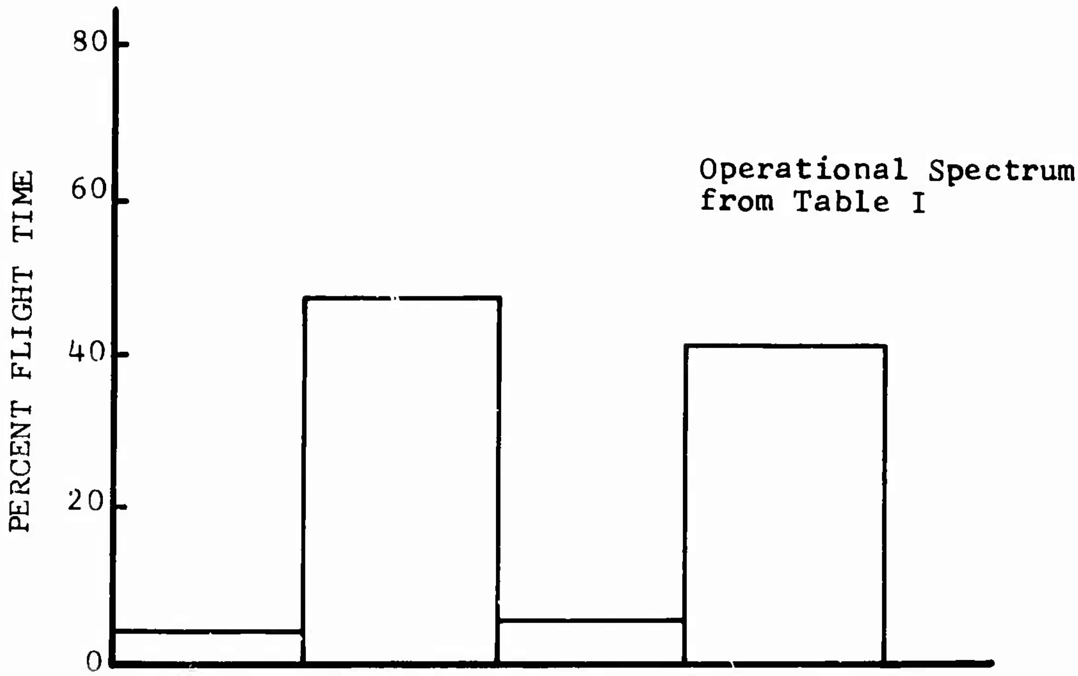
TABLE I. ORIGINAL AND OPERATIONAL AH-1G FREQUENCY-OF-OCCURRENCE SPECTRUMS

| Condition | | Percent Time Original | Percent Time Operational |
|---|---|---|---|
| Ground conditions | | | |
| Normal start | | 0.5000 | 0.4000 |
| Shutdown w/coll | | 0.5000 | 0.4000 |
| IGE Maneuvers | | | |
| Takeoff | | | |
| Normal | | 0.9000 | 1.2780 |
| Jump | | 0.1000 | 0.1420 |
| Hovering | | | |
| Steady | | 2.1700 | 2.0000 |
| Right turn | | 0.1000 | 0.1670 |
| Left turn | | 0.1000 | 0.1670 |
| Control reversal | | | |
| Longitudinal | | 0.0100 | 0.0167 |
| Lateral | | 0.0100 | 0.0167 |
| Rudder | | 0.0100 | 0.0167 |
| Sideward Flight | | | |
| To the right | | 0.2500 | 0.2404 |
| To the left | | 0.2500 | 0.2404 |
| Rearward flight | | 0.2500 | 0.2404 |
| Acceleration | | | |
| Hover to climb A/S | | 0.5000 | 0.5000 |
| Deceleration | | | |
| Normal | | 0.7000 | 0.5000 |
| Quick stop | | 0.3000 | 0.1000 |
| Approach and landing | | 1.0000 | 5.5095 |
| Forward level flight | | | |
| Airspeed | RPM | | |
| 0.50 VH | 314 | 0.5000 | 0.2605 |
| | 324 | 4.5000 | 2.3450 |
| 0.60 VH | 314 | 0.2000 | 0.7723 |
| | 324 | 1.8000 | 6.9508 |
| 0.70 VH | 314 | 0.3000 | 0.8551 |
| | 324 | 2.7000 | 7.6963 |
| 0.80 VH | 314 | 1.5000 | 1.3773 |
| | 324 | 13.5000 | 12.3961 |
| 0.90 VH | 314 | 2.5000 | 0.3990 |
| | 324 | 22.5000 | 3.5910 |
| VH | 314 | 1.0000 | 0.3460 |
| | 324 | 9.0000 | 3.1140 |

| TABLE I - Continued | | |
|---|---|---|
| Condition | Percent Time Original | Percent Time Operational |
| Nonfiring maneuvers | | |
| Full power climb | | |
| Normal | 4.0000 | 2.5000 |
| High-speed | 1.0000 | 0.0426 |
| Maximum rate accel | | |
| Climb - cruise A/S | 2.8000 | 4.6760 |
| Normal turns | | |
| To the right | | |
| 0.5 VH | 1.10000 | 1.6700 |
| 0.7 VH | 1.0000 | 1.6700 |
| 0.9 VH | 2.0000 | 0.1086 |
| To the left | | |
| 0.5 VH | 1.0000 | 1.6700 |
| 0.7 VH | 1.0000 | 1.6700 |
| 0.9 VH | 2.0000 | 0.1086 |
| 0.9 VH control reversal | | |
| Longitudinal | 0.5000 | 0.0835 |
| Lateral | 0.0500 | 0.0835 |
| Rudder | 0.0500 | 0.0141 |
| Sideslip | 0.5000 | 0.2000 |
| Part power descent | 2.5500 | 0.1000 |
| Gunnery maneuvers | | |
| Firing in a hover | 0.0750 | 0.1252 |
| Strafing in accel. from a hover | 0.0500 | 0.0835 |
| Gunnery runs | | |
| PT. Target dives | | |
| To 0.6 VL | 0.2800 | 0.4676 |
| To 0.8 VL | 0.8400 | 2.6003 |
| To 0.9 VL | 1.4000 | 8.0508 |
| To VL | 0.2800 | 0.0200 |
| Spray fire dives | | |
| To 0.6 VL | 0.1200 | 0.2004 |
| To 0.8 VL | 0.3600 | 2.6974 |
| To 0.9 VL | 0.6000 | 5.7264 |
| To VL | 0.1200 | 0.1000 |
| Gunnery run pullup | | |
| To the right | | |
| 0.6 VL | 0.1000 | 0.0500 |
| 0.8 VL | 0.3000 | 0.1000 |
| 0.9 VL | 0.5000 | 0.2500 |
| To the left | | |
| 0.6 VL | 0.1000 | 0.0500 |
| 0.8 VL | 0.3000 | 0.1000 |

| TABLE I - Continued | | |
|---|---|---|
| Condition | Percent Time Original | Percent Time Operational |
| To the left (Cont'd) | | |
| 0.9 VL | 0.5000 | 0.2500 |
| VL | 0.1000 | 0.1670 |
| Symmetrical | | |
| 0.6 VL | 0.0100 | 0.0050 |
| 0.8 VL | 0.0300 | 0.0501 |
| 0.9 VL | 0.0500 | 0.0835 |
| VL | 0.0100 | 0.0167 |
| Gunnery Turns | | |
| To the right | | |
| 0.5 VH | 0.3750 | 0.6262 |
| 0.7 VH | 0.3750 | 1.8236 |
| 0.9 VH | 0.7500 | 0.0400 |
| To the left | | |
| 0.5 VH | 0.3750 | 0.6262 |
| 0.7 VH | 0.3750 | 1.8236 |
| 0.9 VH | 0.7500 | 0.0400 |
| S-Turns | | |
| At 0.8 VH | 0.2000 | 0.1000 |
| At VH | 0.0750 | 0.1282 |
| Power transitions | | |
| Power to auto | | |
| 0.5 VH | 0.0500 | 0.0835 |
| 0.7 VH | 0.1250 | 0.2087 |
| 0.9 VH | 0.1750 | 0.1500 |
| Auto to power | | |
| In ground effect | 0.1500 | 0.1000 |
| 0.4 VH | 0.1000 | 0.1670 |
| 0.6 VH | 0.0750 | 0.8104 |
| Max auto A/S | 0.0250 | 0.0300 |
| Autorotation | | |
| Stabilized flight | | |
| 0.4 VH | 0.2000 | 0.1086 |
| 0.6 VH | 1.4000 | 2.0236 |
| Max auto A/S | 0.3000 | 0.1000 |
| Auto turns | | |
| To the right | | |
| 0.4 VH | 0.0500 | 0.0835 |
| 0.6 VH | 0.4000 | 1.3532 |
| Max auto A/S | 0.0500 | 0.0500 |

| TABLE I - Continued | | |
|---|---|---|
| Condition | Percent Time Original | Percent Time Operational |
| To the left | | |
| 0.4 VH | 0.0500 | 0.0835 |
| 0.6 VH | 0.4000 | 1.3532 |
| Max auto A/S | 0.0500 | 0.0500 |
| Auto Landing | 0.2500 | 0.1000 |



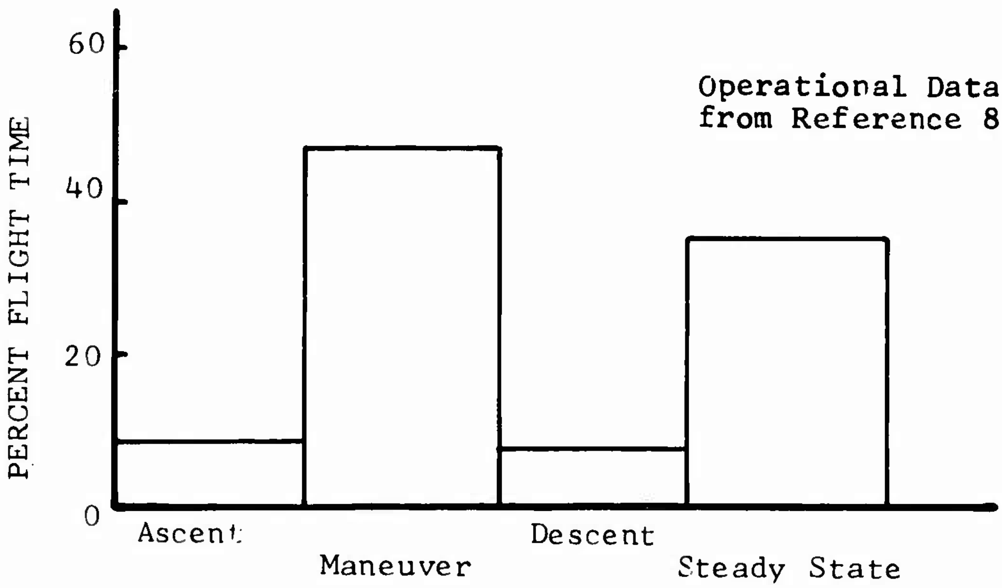




Figure 5. Distribution of Flight Time in Mission Segments for the Operational AH-1G Spectrum and the Operational Data.

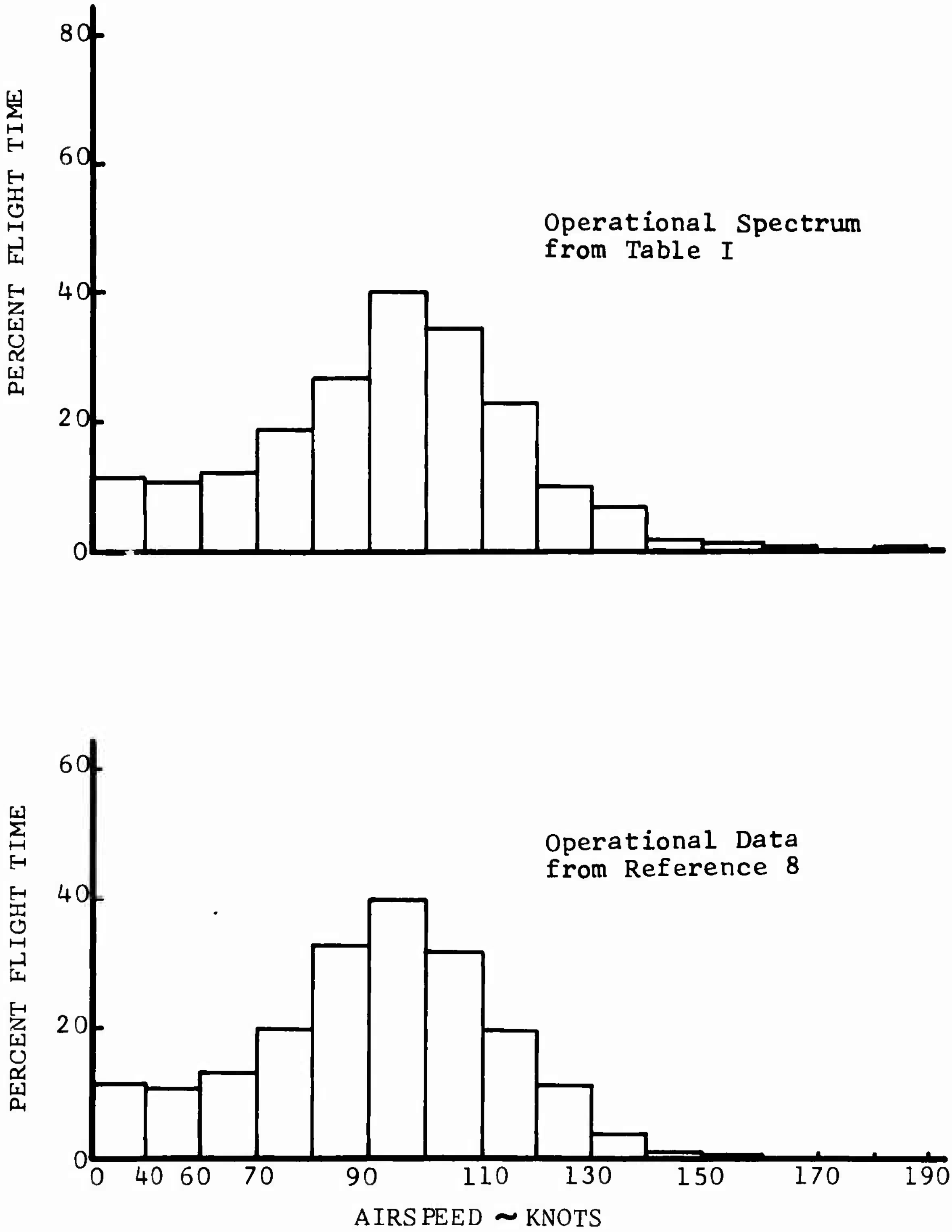



**Figure 6.** Comparison of Airspeed Distribution for the Operational AH-1G Spectrum and the Operational Data.

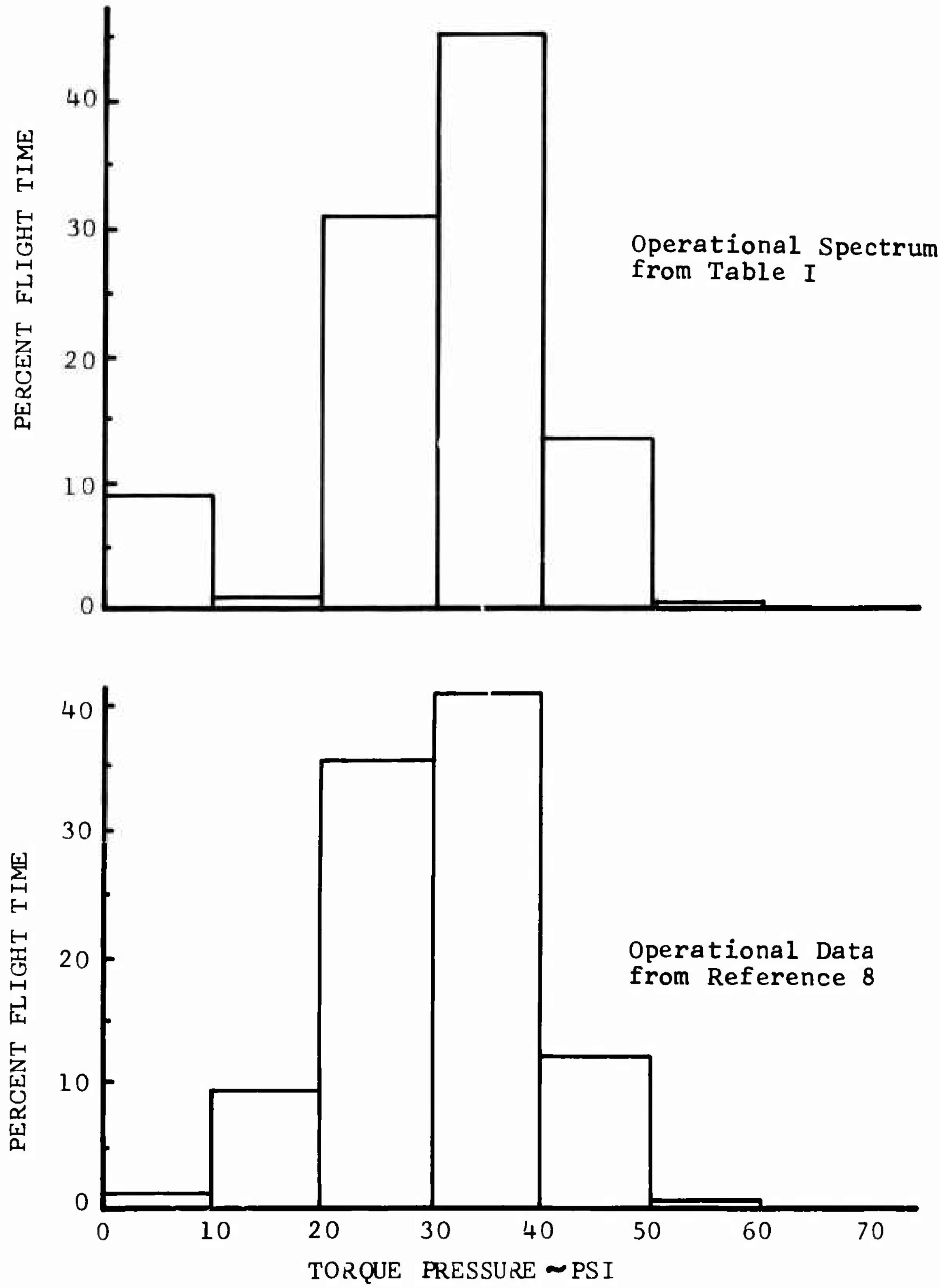



**Figure 7.** Comparison of Torque Pressure Distribution of the Operational Spectrum and the Operational Data.

Rotor speed data are presented in Figure 8. However, from Figure 8 it can be seen that the derived operational spectrum data show similar distributions compared to the operational data in the intervals outside the normal operating range. Therefore, it is felt that the derived operational spectrum adequately represents the operational data with respect to rotor speed.

Figure 9 is included in this report to show the range of vertical load factors for both the derived operational spectrum and the operational data. It is noted that the ordinate for the graph of operational spectrum data from Table I is expressed in percentage of maneuver time while the operational data from Reference 8 is presented in terms of percentage of total occurrences. This difference in presentation was necessary due to the "one measurement per flight condition" procedure followed in reducing the data for the operational spectrum. This procedure precluded any distribution based on occurrences. The data shown for the operational data was obtained by converting "time to reach or exceed" data from Reference 2 into occurrences per 100 flight hours from which the included distribution was calculated. Therefore, this data cannot be considered conclusive. The figure does show that the extreme g levels are adequately represented in the operational frequency-of-occurrence spectrum.

In view of these comparisons of pertinent parameters, the operational frequency-of-occurrence spectrum is considered to represent the AH-1G combat operational data with reasonable accuracy.

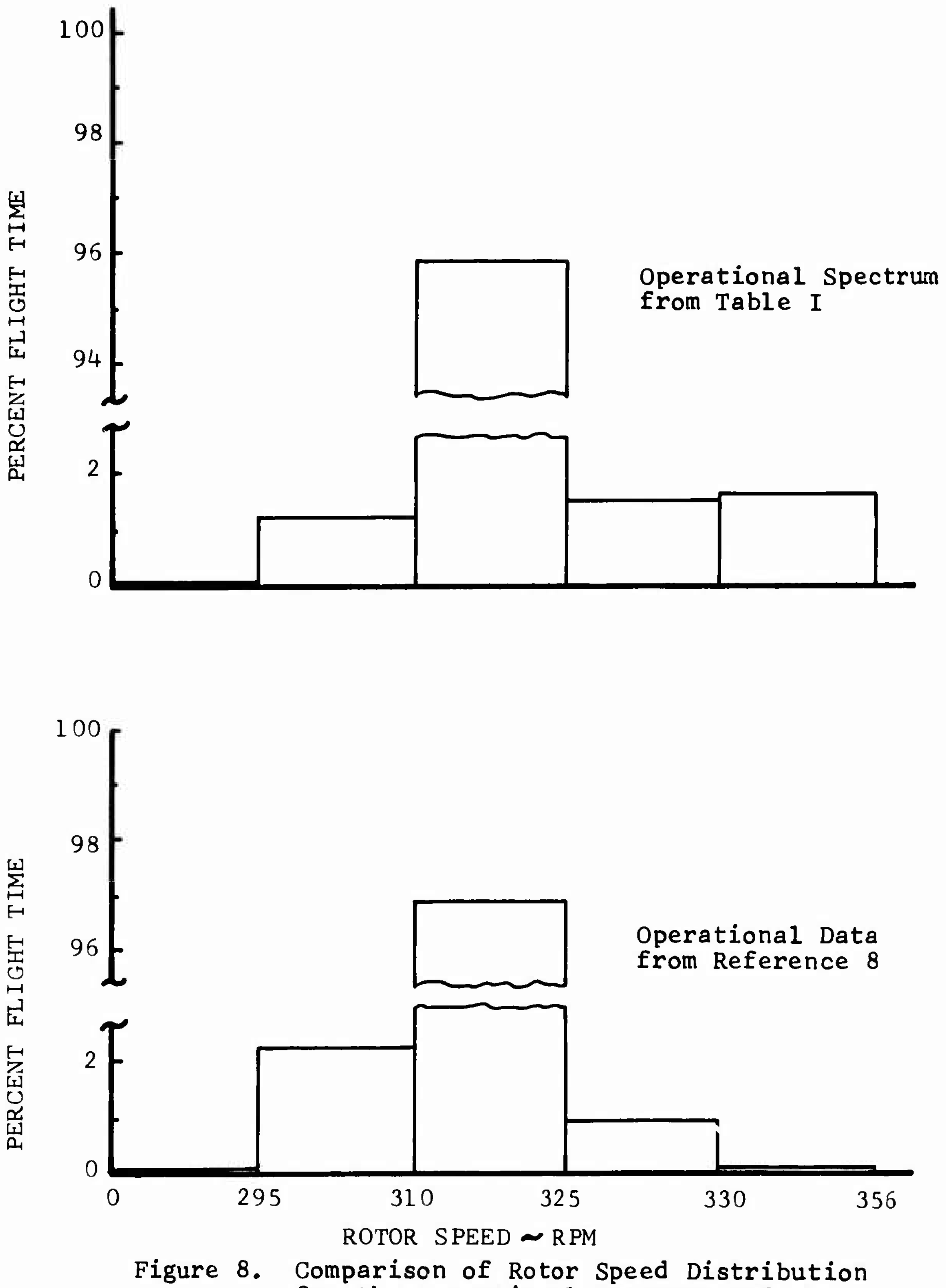



Figure 8. Comparison of Rotor Speed Distribution for the Operational Spectrum and the Operational Data.

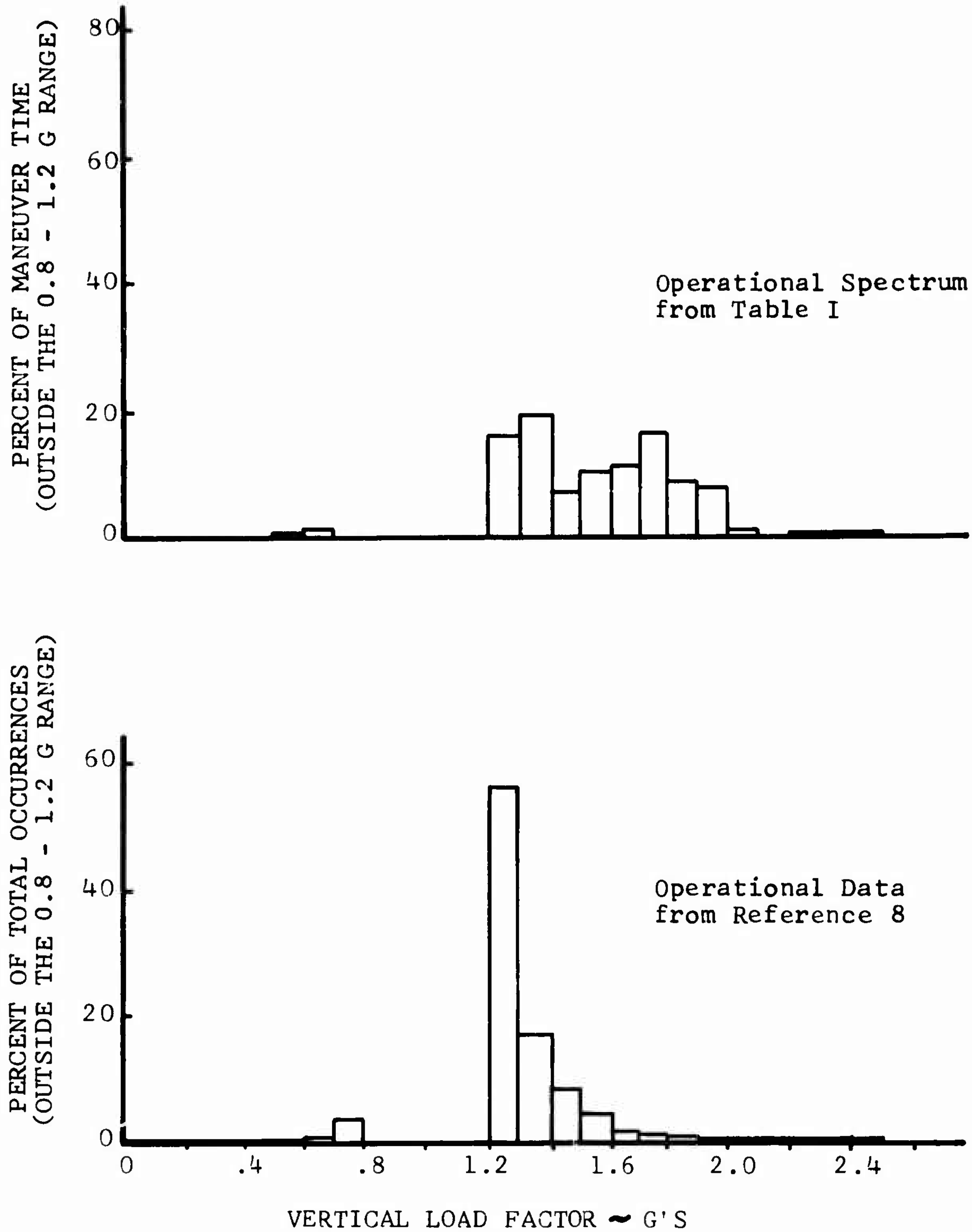



**Figure 9.** Comparison of Vertical Load Factor Distribution for the Operational Spectrum and the Operational Data.

## ADDITIONAL SPECTRUM EVALUATION

The helicopter structural design requirements outlined by the United States Navy are presented in AR-56 (Reference 10). One of the guidelines contained in these requirements is the table of mission profiles for each category of helicopter in the Navy's inventory. The mission profile for attack helicopters, presented in AR-56, has been compared with the operational AH-1G spectrum presented in Table I.

Table II shows the AR-56 spectrum. The frequency of occurrence for most conditions was shown in percentage of total flight time. However, several flight conditions were shown in terms of the number of occurrences per 100 flight hours instead of percentage of time These occurrences are in addition to the other conditions which already total 100 percent of the flight time. Before the AR-56 spectrum could be compared with the modified AH-1G spectrum, these occurrences had to be converted into percentages of total time. To achieve this conversion, appropriate elapsed time in seconds was assigned to each maneuver condition to be changed; see Table III. Using these elapsed maneuver times, the occurrences were converted into total time allotted per 100 hours. These times were then converted into percentage of total time on a 100-hour basis and incorporated into the AR-56 spectrum with corresponding reductions made in the appropriate steady-state condition times so that the summed frequency of occurrence would total 100 percent flight time. Two other adjustments were also made. The 115 percent $V_H$ forward level flight is considered incorrect by the definition of the term $V_H$. Considering $V_H$ to be the forward level-flight airspeed at rated engine power, any airspeed greater than $V_H$ would by necessity be in a dive attitude. Therefore, the time allowed to 115 percent $V_H$ forward level flight was added to the gunnery dive condition. Also, the ground-air-ground cycles were omitted from consideration. They are of a low-cycle nature and are generally accounted for by fatigue test methods and/or analysis during fatigue life substantiation.

Table IV shows the resulting AR-56 spectrum. Two areas of comparison are available between the AR-56 spectrum and the operational AH-1G spectrum shown in Table I. These are distribution of flight time into the four mission segments, and the distribution of forward level-flight time between the various airspeed ranges. Figure 10 shows the mission segment distribution. The obvious difference seen in this distribution is the time allocated to steady-state and maneuver segments. The AR-56 spectrum shows more time in steady state and less time in maneuvers than does the operational AH-1G spectrum. The AR-56 distribution is similar to the original AH-1G spectrum.

TABLE II. ORIGINAL AR-56 SPECTRUM (FROM REFERENCE 10)

| Condition | Percentage of Flight Time |
|---|---|
| On ground | 1.0 |
| Takeoff | (400) |
| Steady hovering | 5.0 |
| Turns hovering | (400) |
| Control reversals hovering | (400) |
| Sideward flight | 1.0 |
| Rearward flight | 0.5 |
| Landing approach | (500) |
| Forward level flight: | |
| 20% VH | 2.5 |
| 40% VH | 4.0 |
| 50% VH | 4.0 |
| 60% VH | 8.0 |
| 70% VH | 8.0 |
| 80% VH | 15.0 |
| 90% VH | 15.0 |
| VH | 15.0 |
| 115% VH | 1.0 |
| Takeoff power climb | 1.0 |
| Full power climb | 3.0 |
| Partial power descents | (500) |
| Power dives | 1.0 |
| Right turns | 3.5 |
| Left turns | 3.5 |
| Control reversals | (2000) |

TABLE II - Continued

| Condition | Percentage of Flight Time |
|---|---|
| Pullups | (500) |
| Power to autorotation | (100) |
| Autorotation to power | (100) |
| Autorotation - steady | 2.0 |
| Autorotation - left turn | 0.5 |
| Autorotation - right turn | 0.5 |
| Autorotation - control reversals | 0.3 |
| Autorotation - landing | 0.3 |
| Autorotation - pullups | (100) |
| Ground air-ground cycles | (100) |
| Gunnery maneuvers: | |
| Hovering | 0.1 |
| Dives | 1.5 |
| Dive pull-outs | 0.65 |
| Turns | |
| Right | 1.0 |
| Left | 1.0 |
| S | 0.15 |

TABLE III. ADJUSTMENTS TO AR-56 SPECTRUM

| Flight Condition | No. of Occurrences per 100 Hours | Elapsed Time (sec) | Time per 100 Hours (sec) | Percentage of Flight Time |
|---|---|---|---|---|
| Takeoff | 400 | 8 | 3200 | .89 |
| Hover turns | 400 | 8 | 3200 | .89 |
| Hover cont. rev. | 400 | 3 | 1200 | .34 |
| Landing approach | 500 | 15 | 7500 | 2.08 |
| Partial power descent | 500 | 15 | 7500 | 2.08 |
| Control reversals | 2000 | 3 | 6000 | 1.67 |
| Pullups | 500 | 7 | 3500 | .97 |
| Power to autorotation | 100 | 2 | 200 | .06 |
| Autorotation to power | 100 | 5 | 500 | .14 |
| Autorotation pullups | 100 | 5 | 500 | .14 |

| TABLE IV. ADJUSTED AR-56 SPECTRUM | |
|---|---|
| Condition | Percentage of Flight Time |
| On ground | 1.0 |
| Takeoff | .89 |
| Steady hovering | 3.77 |
| Turns hovering | .89 |
| Control reversals hovering | .34 |
| Sideward flight | 1.0 |
| Rearward flight | 0.5 |
| Landing approach | 2.08 |
| Forward level flight: | |
| 20% VH | 1.61 |
| 40% VH | 1.92 |
| 50% VH | 1.92 |
| 60% VH | 8.0 |
| 70% VH | 8.0 |
| 80% VH | 12.36 |
| 90% VH | 15.0 |
| VH | 15.0 |
| Takeoff power climb | 1.0 |
| Full power climb | 3.0 |
| Partial power descents | 2.08 |
| Power dives | 1.0 |
| Right turns | 3.5 |
| Left turns | 3.5 |
| Control reversals | 1.67 |

TABLE IV - Continued

| Condition | Percentage of Flight Time |
|---|---|
| Pullups | .97 |
| Power to autorotation | .06 |
| Autorotation to power | .14 |
| Autorotation - steady | 1.66 |
| Autorotation - left turn | 0.5 |
| Autorotation - right turn | 0.5 |
| Autorotation - control reversals | 0.3 |
| Autorotation - landing | 0.3 |
| Autorotation - pullups | .14 |
| Gunnery Maneuvers: | |
| Hovering | 0.1 |
| Dives | 2.5 |
| Dive pull-outs | 0.65 |
| Turns | |
| Right | 1.0 |
| Left | 1.0 |
| S | 0.15 |

Figure 11 shows the forward level-flight airspeed distributions for both spectrums. The AR-56 spectrum shows a larger amount of time at the higher airspeeds and less in the mid-range airspeeds. Again, this is similar to the original AH-1G spectrum.

In general, the AR-56 spectrum is much closer to the original AH-1G spectrum than to the operational spectrum. This is very likely a result of the Navy's using the frequency-of-occurrence spectrums for the UH-1E and AH-1J, which are very similar to the original AH-1G spectrum, as a basis for the preparation of the spectrum contained in AR-56.

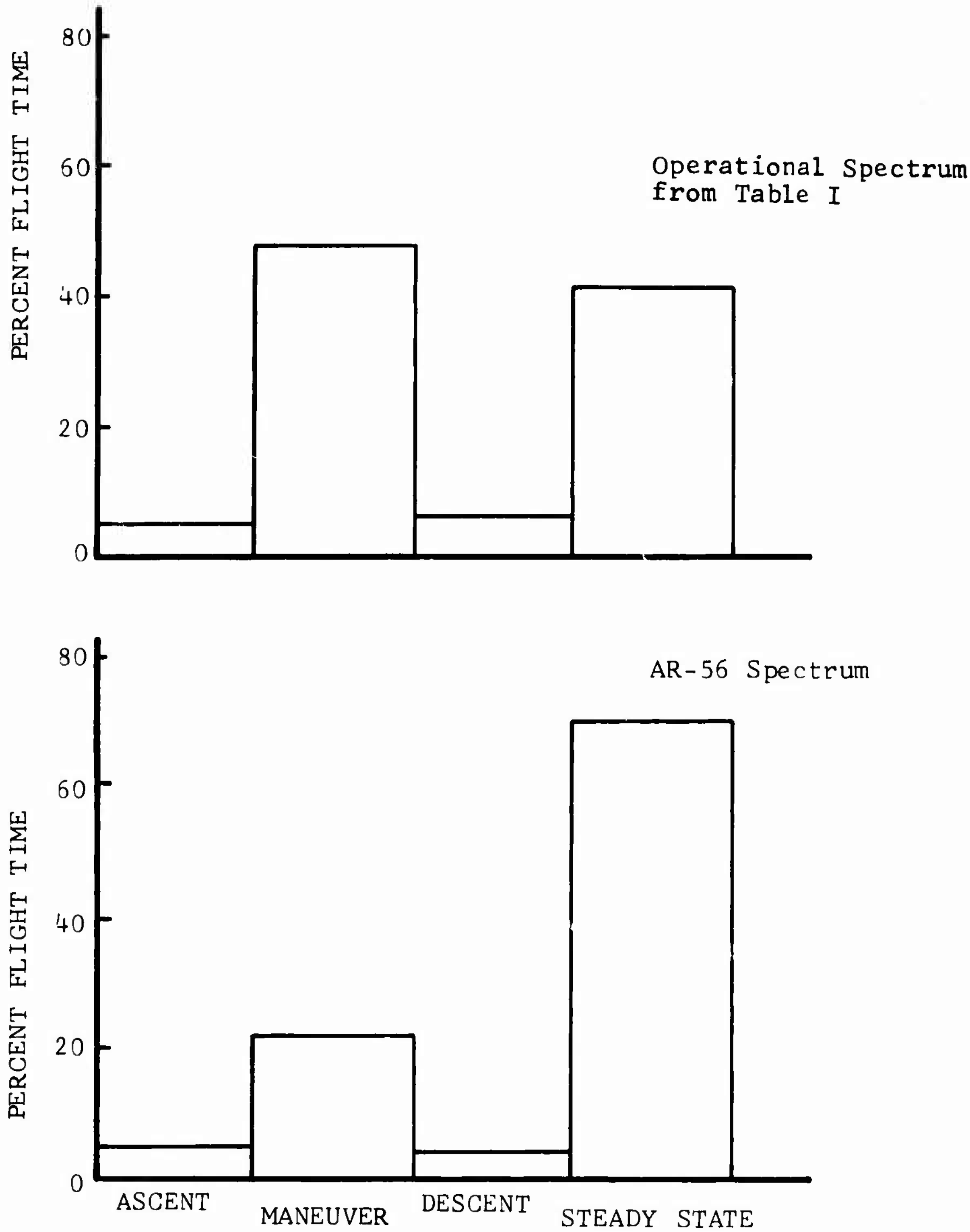



Figure 10. Comparison of Mission Segment Distribution of the Operational Spectrum and the AR-56 Spectrum.

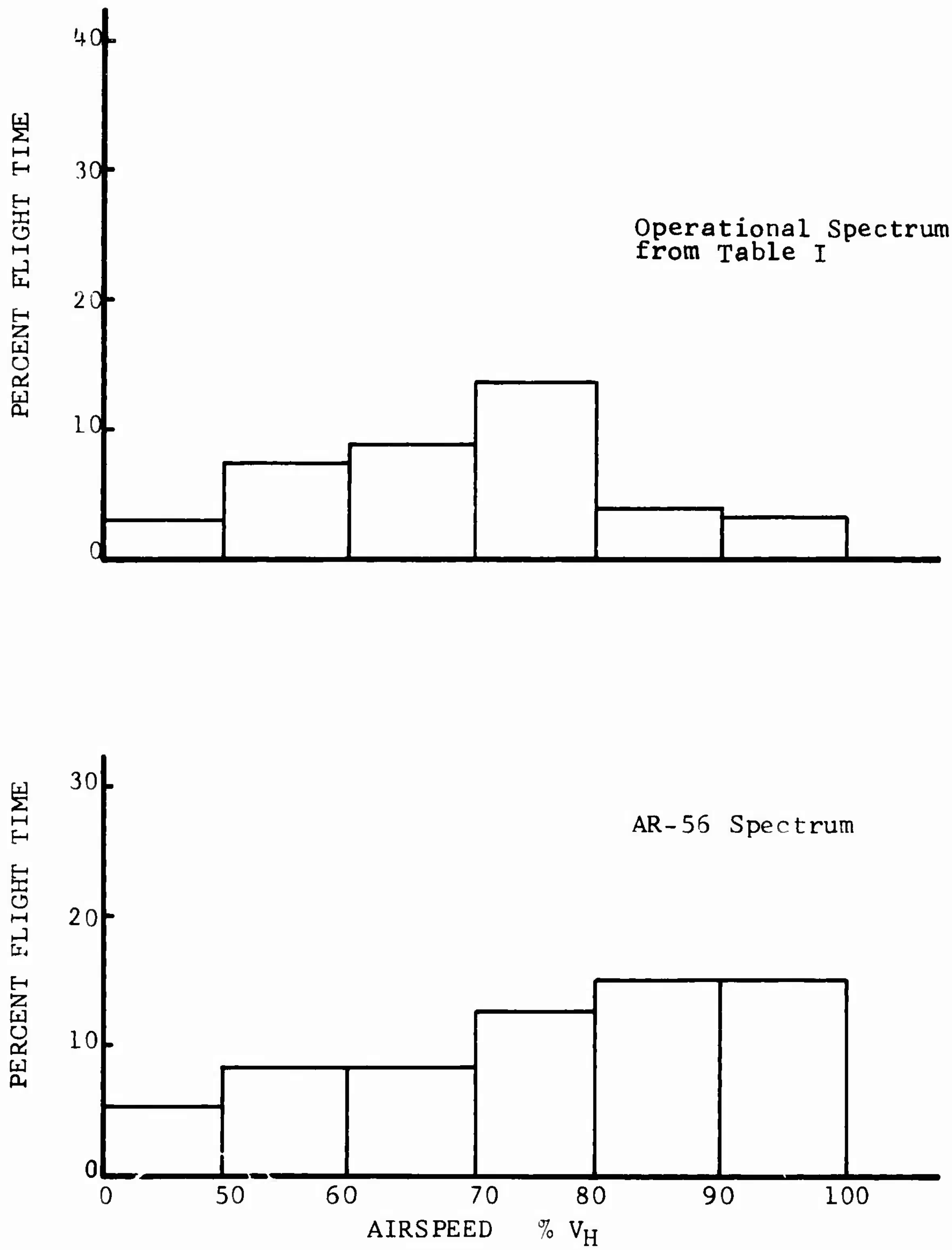



Figure 11. Comparison of Forward Level-Flight Airspeed Distribution for the Operational Spectrum and AR-56 Spectrum.

## FATIGUE LIFE CALCULATIONS

Using the operational frequency-of-occurrence spectrum, fatigue lives were calculated for the following AH-1G dynamic components: main rotor blade, main rotor grip, main rotor yoke extension, swashplate outer ring, tail rotor blade, and tail rotor grip. These components were chosen since they are representative of the component categories of main rotor hub and blade, tail rotor hub and blade, and main rotor controls. Also, there are parts of both ferrous and nonferrous materials within this group. For these reasons, any significant effects caused by the spectrum modification should be evident in the fatigue life calculations. The fatigue life calculations shown in the Appendix are summarized in Table V.

Table V also lists the fatigue lives based on the original frequency-of-occurrence spectrum. The effects of the spectrum modification were neither totally detrimental nor totally advantageous. The component lives which showed the most change were those with only a few damaging conditions such as the main rotor yoke extension and the swashplate outer ring. The other component lives showed very little change due to the spectrum modification.

Although there were some relatively large modifications in some areas of the spectrum, the net effect on the resulting fatigue life was not significant. This is attributed to the fact that even though some of the individual changes seemed rather drastic, they tended to compensate for each other. The change in mission segments, wherein the time spent in level flight was reduced and the time spent in maneuver was increased, would lead one to expect a reduction in fatigue life. However, this was apparently compensated for by the reduction in severity of the airspeed distribution, since the oscillatory loads are strongly dependent upon airspeed. The resulting differences in fatigue lives do not appear to be sufficient to warrant a change in recommended retirement intervals.

| TABLE V. SUMMARY OF AH-1G FATIGUE LIVES | | |
|---|---|---|
| Component | Original Spectrum (hour) | Operational Spectrum (hour) |
| Main rotor blade | 2,792 | 2,476 |
| Main rotor grip | 95,057 | 45,454 |
| Main rotor yoke extension | 10,633 | 9,334 |
| Swashplate outer ring | 9,806 | 19,443 |
| Tail rotor blade | 3,764 | 4,827 |
| Tail rotor grip | 8,103 | 7,587 |

# 540 ROTOR SYSTEM DEVELOPMENT AND FATIGUE DESIGN METHODOLOGY

This section reviews the 540 rotor system development from inception to present use on the AH-1G helicopter. The fatigue design methods employed in this time period are reviewed, and where service experience indicated that a fatigue design objective was not met, the reason for the design shortcoming is discussed.

## DEVELOPMENT CYCLE

A chronological listing of the significant events of the 540 rotor system design-development-use cycle is shown in Table VI. The objective of the initial design was to develop a rotor system which would give a smoother ride at high speeds, require less maintenance, and have a higher rotational inertia than the UH-1 rotor then in production. The rotational inertia requirement was met by using large tip weights in the blade. Since dynamic considerations in two-bladed semirigid rotor systems require that the first in-plane frequency be well above the rotor operating speed, the increased tip weights made it necessary to provide a corresponding increase in the in-plane stiffness of the rotor system. This was accomplished by using an increased chord blade and by employing a unique door-hinge concept in the feathering bearing region of the rotor hub. Compared with the more conventional spindle-housing bearing arrangement, the fore and aft placement of the feathering and nonfeathering elements of the door-hinge hub provide a more effective utilization of material in obtaining an in-plane stiff structural arrangement. To further stiffen the feathering bearing region, the spanwise spacing of the bearings was increased to approximately twice that used in previous designs. This increased spacing also served to reduce bearing loads and thus increase bearing life. Inboard of the hub feathering bearings, a wide flat-plate structure was used to provide high in-plane stiffness and vertical softness. This flexural plate element served to reduce vertical bending moments in this area by acting as a virtual hinge.

The reduced vibration levels projected for the design were confirmed in an experimental flight test program. In this program, the prototype 540 rotor system was flown on a Model 204B helicopter. Following an evaluation by the Army Test and Evaluation Command, the 540 rotor system was considered for application to the UH-1 series helicopters.

A production design for the UH-1B was initiated. This design retained the basic rotor system geometry of the prototype, and

TABLE VI. 540 ROTOR SYSTEM DESIGN-DEVELOPMENT-USE CYCLE

| | |
|---|---|
| 1962 | Initial design |
| 1963 | Experimental flight test<br>UH-1B/1C design |
| 1964 | Evaluation by U.S. Army Test and Evaluation Command<br>UH-1C production design<br>Flight test and fatigue test |
| 1965 | Production delivery UH-1C<br>AH-1G design |
| 1966 | - |
| 1967 | Production delivery AH-1G |
| 1968 | Initiate operational flight surveys |
| 1969 | Main rotor blade modification |
| 1970 | Complete operational flight surveys |

changes were limited to those typically involved in going from experimental-type hardware to production fabrication. Following flight test and fatigue test, the first production delivery of a 540 rotor system on a UH-1B airframe, now denoted as the UH-1C, was made in June 1965.

In 1965, a program was initiated to develop a high-speed attack-type helicopter. This helicopter, denoted as the Model 209, would use the 540 rotor system. The helicopter features a slim fuselage design; and as a part of the drag reduction program, the stabilizer bar used with the UH-1C control system was discarded in favor of an electronic stabilization system. This resulted in some minor revisions to the components in the dynamic control system. Production delivery of the AH-1G (formerly the Model 209) started in May 1967. One fatigue-oriented structural modification of the rotor system was required subsequent to delivery. This involved a main rotor blade modification and is discussed in more detail later in this section.

Two operational flight surveys of helicopters using the 540 rotor system were conducted from 1968 to 1970. One of those, Reference 11, provided data on the UH-1C and AH-1G operations in Southeast Asia. The second, Reference 8, provides operational data on the AH-1G and is the source of information on which this report is based.

## FATIGUE DESIGN METHODS

In the design of fatigue-loaded components, a relationship between the fatigue strength and the magnitude of the flight loads must be established. At different stages in the design-development cycle, the manner in which a fatigue analysis is performed will vary. During the design stages, fatigue strength data from previous component tests and calculated flight loads are used. During the development phase, components are fabricated and flown, flight loads are measured, and components are fatigue tested. At this time, flight loads and fatigue test data on the actual components are used to estimate the relationship between fatigue strength and flight loads. Finally, the component enters service and demonstrates through satisfactory operation or premature failures the true relationship between fatigue strength and flight loads. The common measure of this relationship is fatigue life.

In the preliminary design stage, three methods are commonly used to size components for fatigue loading. The first method is to calculate flight loads for each discrete flight condition, to estimate the frequency of occurrence for each condition, to estimate the component's endurance limit, and to calculate a fatigue life for the part. Several problems arise when this approach is applied to a practical situation: (1) accuracy is poor in calculating oscillatory loads in maneuvers, (2) the frequency-of-occurrence estimate has to be made before adequate mission and performance data have been obtained, and (3) the calculation process is lengthy and must be repeated each time the structural configuration is changed. Since a considerable number of such changes are usually made during the design stages, this type of analysis can become inefficient and impractical.

A second, far less sophisticated method involves the use of an additional factor applied to the static design loads to ensure low working stresses under actual loading conditions. In this method, a static analysis only is performed and fatigue considerations are included by using a factor which in effect is the ratio of the ultimate or yield strength to the endurance limit. While this method is sometimes used in the analysis of secondary structure or in redundant airframe structure, it is not considered adequate for application to dynamic components.

The third method, developed from experience, utilizes dynamic loads for a single key flight condition as the basis for the fatigue evaluation. Prior to the initial 540 rotor system design effort, it had been shown that for a typical helicopter loading spectrum, a satisfactory fatigue life would be obtained for an aluminum part when the endurance limit of the part was at or just above the alternating stresses for maximum level-flight airspeed ($V_H$). For steel components, it was shown that the flatter shape of the S-N curve made it necessary to keep the endurance limit approximately 40 percent above the $V_H$ level-flight stresses to obtain a satisfactory life. In this method, the need for accurate maneuver loads calculation is thus circumvented by the use of these empirical relationships. This method was used in the preliminary design of the 540 rotor system.

## FATIGUE STRENGTH ALLOWABLES

When designing a new component, the fatigue strength is usually determined by using the results of prior fatigue tests of full-scale, geometrically-similar parts and material. In the case of the 540 rotor system, fatigue data of similar parts were available for most major components. The fatigue strength of the components was, therefore, predicted with a relatively high degree of confidence at the preliminary design level. The same fatigue strength allowables were used later for sizing the production-design rotor components.

Since fatigue tests of all critical rotor components were to be conducted prior to production delivery of the UH-1C, the philosophy used was to design for light weight and modify later as indicated by fatigue test results. With the exception of two joint areas, all components exhibited fatigue strengths equal to or greater than the predicted values used in design. It was only necessary to provide moderate local increases in some sections to obtain the desired fatigue strengths.

When the 540 rotor was considered for use on the Model 209 Cobra, the fatigue strenghts established in the UH-1C tests were used in the fatigue analysis. As mentioned previously, the Model 209 dynamic control system was newly designed to eliminate the stabilizer bar. In the design stage, fatigue strengths of these new components were determined from prior fatigue tests conducted on parts of similar geometry and material. Subsequent testing of the new control system verified that the predicted component strengths were equal to or greater than the predicted values used in the design phase.

## FATIGUE DESIGN LOADS

At the time of the preliminary design of the 540 rotor system, the available analytical tools for load calculations were far less sophisticated than those in use today. Oscillatory rotor loads for only the steady-state level flight could be calculated with reasonable accuracy. At this point in time, it was, therefore, customary to place more confidence in empirical methods to determine the steady-state level-flight fatigue design loads and to use the analytical methods to study the effects of design variations on rotor loads.

To establish the fatigue design loads for the 540 rotor in the preliminary design phase, flight loads measured in test programs of other two-bladed rotor systems were used. It was found that reasonable correlation of the oscillatory rotor loads of different helicopter models could be achieved when they were compared on a thrust-coefficient and advance-ratio basis. While this method, which considers only aerodynamic parameters, did not account for the differences in dynamic characteristics of the rotor systems, it was found that for the two-bladed, semirigid, teetering rotor family, these differences were small. This empirical method was used in establishing the $V_H$ level-flight fatigue design loads for the initial 540 rotor design.

For the 540 production design, the $V_H$ level-flight loads measured in the flight testing of the prototype rotor were used. Since the complete envelope of steady-state and maneuver flight had been investigated in the flight test program, these design loads were used with a high degree of confidence.

By the time the 540 rotor was considered for the AH-1G, a considerable amount of testing and service experience with the UH-1C had been accumulated. While the airspeed of the AH-1G was to be higher than that of the UH-1C, loads calculations showed the main rotor fatigue loads for the $V_H$ level-flight condition to be about the same, because the airspeed increase was made by a reduction in fuselage drag and download. The power available, a parameter which is strongly related to oscillatory rotor loads, was the same for both the AH-1G and the UH-1C.

## EVALUATION OF FATIGUE DESIGN/ANALYSIS METHODS

The true fatigue life of helicopter dynamic components is proven only after the aircraft has been in service and has accumulated a significant number of hours operating under the various actual mission conditions. In the case of the 540 rotor, a considerable history of operation is available for this system on two different helicopter models: the UH-1C and the AH-1G. A review of this record shows that of the 25 fatigue-loaded components of the main rotor and control system, only one significant design modification was required due to fatigue problems which were encountered in service. This change involved a redesign of the main rotor blade which was incurring premature fatigue failures of the blade skin at approximately mid-span. While the failures were demonstrated to be of a fail-safe mode, it was felt that the problem could be eliminated by a relatively small design change which would be economically attractive. In evaluating the fatigue design methodology, it is pertinent to examine this main rotor blade failure to determine the cause and to identify any shortcomings in the analysis which may have prevented the original design from reaching the fatigue design goal.

The cause of the blade failures was traced to a high-frequency oscillatory load which was present over a rather narrow band of the level-flight airspeed envelope. A combination of blade frequency placement and airloads due to trailing tip vortices caused the blade to be excited at 6/rev at a level-flight airspeed of approximately 90 knots. Since the problem was associated with the blade passage frequency through the shed vortices, the excitation was airspeed-sensitive, and the resonant condition disappeared at speeds above or below the critical speed. Because of the relatively small size of this critical band, the significance of its presence and magnitude was not fully realized during the flight-test program. Subsequent analyses conducted to establish the retirement life of the blade showed the main blade retention joint rather than the mid-span station to be the most fatigue-critical area. It is interesting to note that this condition was present at an airspeed which was assigned a relatively small percentage of time in the original engineering frequency-of-occurrence spectrum. As shown in Figure 4, the larger percentages of time in level flight are assigned to the higher airspeeds. While this would usually be conservative, it was not the case in this particular instance. It will also be noted in Figure 4 that the mission profile data show a much larger percentage of time assigned to the 90-knot level-flight condition. While these data were not available at the time the problem was being investigated, they do, in retrospect, help to explain the shorter-than-predicted fatigue life.

The modifications in the blade redesign were directed toward eliminating the high-frequency resonance problem and providing moderate increase in structural capacity of the outboard blade section. This was accomplished by changing the taper of the trailing-edge strip to increase the chordwise section stiffness. In reviewing the fatigue design methods employed in the design of the original 540 rotor system, it is immediately obvious that the method does not anticipate problems of the type encountered in the 540 blade service failures. In the original design, level-flight loads from previously flown two-bladed rotor systems were used to establish fatigue design loads. While these loads were adjusted to account for the differences in aerodynamic parameters, it was assumed that the dynamic characteristics would be similar. This assumption proved reasonable for the lower frequency loadings but was not valid for the particular problem encountered with the high-frequency loads exhibited in the 540 blade.

With the development of more sophisticated rotor loads calculation programs, improvements in dynamic rotor loads prediction have been achieved for the lower frequency excitations. Although programs for predicting the presence and location of resonant frequencies for a particular rotor design are quite accurate, the ability to predict the magnitude of the higher frequency loads accurately is not currently within the state of the art.

Of the three primary variables used to establish rotor component fatigue lives in the preliminary design stage - (1) component fatigue strength, (2) calculated oscillatory loads, and (3) frequency of occurrence - the area of rotor loads calculation appears most deficient. The estimation of component fatigue strength can be made with a relatively high degree of confidence due to the large and ever-growing data base established by full-scale component fatigue testing. The frequency-of-occurrence spectrum takes on major significance in the preliminary design phase only after the ability to calculate accurate maneuver loads is achieved.

Later in the development cycle, when fatigue testing has been completed and measured flight loads are available, the frequency-of-occurrence spectrum becomes the key variable in establishing fatigue lives. It is important, therefore, that the collection of mission profile data be continued so that the most realistic frequency-of-occurrence information can be used in establishing component retirement intervals.

In recent years, the rapid advances in computer technology have been nearly paralleled by rotor loads prediction capability. Refinements, particularly in the calculation of oscillatory loads in maneuver conditions, now make it practical to compute fatigue loads of the total flight spectrum. This method is still time consuming, however, and the empirical fatigue design methods are more economical and convenient to use in the early stages of design iteration. When the design is reasonably firm, the fatigue life is calculated using calculated oscillatory loads for each maneuver condition of the total flight spectrum. The empirical methods then provide a check of this calculation. For this reason, a study was conducted in 1967 to review and update the empirical method which was then in use. The method which evolved from this study, Reference 12, utilizes the previous approach of designing with $V_H$ level-flight stress and the component endurance limit to meet a given fatigue life objective. Refinements were made to consider various materials and the differences in sensitivity of the main rotor, tail rotor, and rotating controls to the maneuver loading spectrum. Also, additional flexibility was added to the method by incorporating a spectrum severity index factor which is dependent on the helicopter mission.

## MAXIMUM ONE-TIME OCCURRENCES AND LIMITATIONS

The purpose of the work in this section is to compare the values of maximum one-time occurrences measured in Reference 8 with those specified in the structural design criteria, measured in the helicopter flight structural demonstration, or contained in the aircraft operating limitations. A study of the various parameters was made in an attempt to determine what limiting factor caused each maximum one-time value.

### COMPARISON OF ONE-TIME VALUES

The maximum one-time values recorded in the Southeast Asia survey for airspeed, normal acceleration, rotor speed, engine torque, and gross weight are shown in Table VII. These are compared with values contained in the aircraft operating limitations, specified in the structural design criteria, and measured in the flight structural demonstration.

A maximum airspeed of 185 knots was measured in the Southeast Asia data. This occurred at an altitude of 5000 feet and at a gross weight of 7546 pounds as the aircraft was pulling out of a descent at 5000 feet per minute. In this altitude range, 3000 feet to 6000 feet, the aircraft is redlined at 174 knots (approximately 1.20 times $V_H$). The pilot was, therefore, exceeding the operating limits of the aircraft. The airspeed was still well below the 222-knot speed used for structural design and the 210-knot dive recorded in the structural demonstration flight test.

The maximum operational rotor speed of 351 rpm, which occurred in autorotation, was also above the aircraft redline value of 339 rpm. This value is lower than the 356 rpm specified in the structural design criteria but higher than the 342 rpm recorded during the structural demonstration. In establishing the 356 rpm as a design limit in the structural design criteria, a deviation from the MIL-S-8609 specification factor of 1.25 on rotor speed was granted. The factor of 1.05, used for the AH-1G and previous designs of the UH-1 series, appears to be realistic for rotor overspeed considerations for the attack helicopter. Although no factor on the low range of rotor speed was included in the structural design criteria, a 1.05 factor was used in the structural analysis of the rotor, Reference 16, to establish a minimum design limit rotor speed of 280 rpm. No discrete value for extreme minimum rotor speed was reported in Reference 8, only that it was less than 295 rpm. The aircraft operating limit is 294 rpm minimum.

No discrete value for maximum engine torque was given in Reference 8. The maximum value recorded was greater than 60 psi and less than 70 psi. This is in excess of the aircraft

TABLE VII. COMPARISON OF AH-1G MAXIMUM ONE-TIME OCCURRENCES

| Parameter | Aircraft Operating Limitations (Ref. 13) | Structural Design Criteria (Ref. 14) | Structural Demonstration Flight Test (Ref. 15) | Southeast Asia Operational Flight Survey (Ref. 8) |
|---|---|---|---|---|
| Airspeed (kt) | SL-3000 ft, 190<br>6000 ft, 174 | 222 | 210 | 185 |
| Rotor speed (rpm) | Power on, 294-324<br>Power off, 294-339 | Power on, 294-356<br>Power off, 294-356 | 284-342 | 351 |
| Engine torque (psi) | 50 | 62.5 | 59 | >60 |
| Gross weight (lb) | 9500 | 6600 9500 | 7500 9400 | 9522 |
| Normal acceleration (g) | Flight at or near zero g prohibited | 3.50 2.43<br>-.50 -.50 | 2.73 2.10 | 2.4<br>0.1 |

redline of 50 psi and may be greater than the 62.5 psi used in the structural design criteria. The maximum value recorded in structural demonstration maneuvers was 59 psi.

The gross weight of 9522 pounds reported in the Southeast Asia survey is slightly above the aircraft operating limitation of 9500 pounds. A maximum gross weight of 9500 pounds is specified in the structural design criteria. The structural flight demonstration was conducted at a maximum gross weight of 9400 pounds. The extreme values of normal acceleration reported in Reference 2 were +2.4g's and +0.1g. The aircraft operating limits do not specify any maximum g limitation, and there is no g-meter in the helicopter instrumentation. The Operator's Manual does specify that aerobatic maneuvers and flight at or below 0g are prohibited. The normal acceleration specified in the structural design criteria ranges from +3.50g's to -.50g. In the structural demonstration flight, a maximum value of 2.75g's was measured.

## OTHER ONE-TIME MEASURED VALUES

In addition to the maximum one-time values of the parameters listed in Table VII, Reference 8 presents measured values for longitudinal and lateral accelerations, collective and cyclic stick positions, density altitude, and outside air temperature. Although limits for these parameters are not listed in the operating limits or structural design criteria, a brief discussion of each parameter is presented.

The maximum lateral acceleration was .45g left and right. The maximum longitudinal acceleration was .25g forward and .30g aft. No lateral or longitudinal accelerations are specified in the design criteria. In putting the helicopter in equilibrium for the various flight conditions, maximum accelerations of 1.33g lateral and .37g longitudinal were determined. These accelerations were used for structural analysis of the airframe.

The extreme forward longitudinal cyclic stick was in the 0 to 10 percent increment. The extreme aft position was in the 70 percent to 80 percent increment. The extremes recorded for collective stick were in the 0 to 10 percent increment for the most down position and in the 70 percent to 80 percent increment for the most up position.

The maximum density altitude measured was above 10,000 feet and less than 15,000 feet. The outside air temperature ranged from a minimum of 40°F to a maximum of 100°F.

## PROBABLE CAUSE OF LIMITING FACTORS

To determine the probable cause of the limits reached in Reference 8, discussions were conducted with people having expertise in various helicopter design disciplines, with test pilots, and with pilots with AH-1G combat experience. Additionally, pertinent literature was surveyed. Limits of the vehicle as seen from the perspective of design, test, and operational personnel provided the composite view necessary to make a balanced evaluation.

These discussions and a review of pertinent literature brought out a point which must be considered. Although the two data samples are an important contribution to the literature and will be used at face value, there is considerable evidence that the pilots flew by the book since they knew they were being monitored. An unbiased sample, if such were possible, would be of even greater value. Since the present report compiles best guesses as to the reasons for the maximum one-time occurrences, comments which indicate data biasing seem important to the explanation and will be included.

Difficulties were encountered in some cases due to the limited presentation of data in Reference 8. When evaluating the cause for the limit of a particular parameter, it would be desirable to have the instantaneous value of all other measured parameters, or even better, a short time history of the event such as presented in Reference 17. This would aid in reconstructing the condition and thus give more credibility to the argument for the cause.

## AIRSPEED

The maximum airspeed of 185 knots was measured at 5300 feet altitude with the aircraft at 7546 pounds gross weight pulling out of a 5000-feet-per-minute descent. This maneuver was not reported as a gunnery run. This odd point can be put in perspective by noting a couple of statistics: (a) less than one minute in 202 hours (<.008%) was spent above 170 knots, and (b) only 60 minutes (.5%) was spent above 140 knots. From a gross perspective, the airspeed data peaked in the 80- to 100-knot range, which is considerably below $V_H$--140 to 150 knots for clean configuration and 130 to 140 knots for HOG (19 tube rocket pods inboard and outboard) configuration.

Vibration and handling qualities are good enough to permit the AH-1G to spend a high percentage of time at $V_H$; thus the peaking of the speed spectrum at a lower value--$0.5V_H$ to $0.75V_H$--is due to factors other than flight characteristics, probably theater tactics.

Further, the AH-1G can extend its speed from the 140-150 knot level-flight range to the redline speed of 190 knots by diving. The data show that high-speed dives were not frequently made. Reasons for this probably include both ship's characteristics and theater tactics.

High-speed dive characteristics vary from mild to severe depending on how the maneuver is executed. Consider the following excerpt which contains most of what is presented in the Operator's Manual, Reference 13.

> "Diving flight presents no particular problems in the AH-1G; however, the pilot should have a good understanding of such things as rate of descent versus airspeed, rate of closure, and rates of descent versus power. Because of relatively low drag, the aircraft gains airspeed quite rapidly in a dive, and it is fairly easy to exceed the redline. Rates of descent of 3500 feet per minute to 4800 feet per minute at full power are not uncommon during high speed dives. These high rates of descent coupled with the high flight path speeds (320 feet per second at 190 knots true airspeed) require that the pilot monitor both rate of closure and terrain features very closely and plan his dive recovery in time to avoid having to make an abrupt recovery. If an abrupt recovery is attempted at airspeeds near redline airspeed, "mushing" of the aircraft can occur. If mushing is experienced, do not increase collective. Application of increased collective will aggravate the condition.
>
> At speeds above the maximum level flight speed, the rate of descent will increase approximately 1000 feet per minute for every 10 knots increase in airspeed for the full power condition. At redline airspeed, the rate of descent will not change appreciably for any torque pressure between 40 and 50 psi."

The excerpt above does not tell all there is to know about a dive and recovery. There is a problem in determining how much information to put in the manual and how much to leave to flight training. Characterizing a dive is difficult since there are so many variables involved: entry speed, power, and altitude; dive angle and power; recovery method and constraints; etc. Most of the limitations are on the recovery, not on the dive itself.

Flight characteristics which deter extreme dives are steep dive angles, reaction time for recovery from engine failure, closure speed, vibration, and recovery constraints. For example, there is little deterrence to a steep dive from 7000 to 3500 feet ground height, but there is extreme deterrence to the same maneuver executed between 4500 and 1000 feet. Vibration is not the principal or even a serious deterrent, although it is a consideration and deserves some discussion. The main fuselage vibrations are at frequencies which are multiples of rotor speed, namely, two-, four-, and six-per-rev. Since lateral vibration is never a problem, only the vertical will be discussed. Cabin vibration comfort at $V_H$ is good but degrades with increasing dive speed, reaching the following maximum values near 180 knots: 2/rev, ±0.3g; 4/rev, ±0.5g; 6/rev, ±0.5g (Reference 18). These are not acceptable levels for steady-state flight, but they are not severe enough to seriously deter high-speed transients which last only a few seconds in a regime where pilots expect high vibration; however, there is a learning curve involved. Pilots accept increased vibration in transients such as high g turns and landing flares if they repeat the maneuver often enough to learn what is normal. A pilot who does not regularly execute high-speed dives will have trouble learning the normal vibration characteristics of the AH-1G since the harmonic content varies with several parameters: loading configuration (gross weight, cg, and stores), airspeed, density altitude, power, and rpm. Additionally, the pilot and gunner will frequently disagree on the effect of a given parameter since the variation in a given harmonic at the two seats is often opposite. These variations are readily explainable in terms of the response mode shapes; however, no conscious effort has been made to teach this pattern to AH-1G flight crews.

The dive recovery varies from routine to severe depending on dive speed and angle, and recovery constraints such as altitude, obstacles, air traffic, and enemy fire. If it is assumed that the high-speed dive is an indispensable tactic, that it will be used frequently, and that recoveries will have to be made under the most extreme conditions, then there is no substitute for rigorous instruction and drill in making high-speed dives and recoveries during flight training. In the field, practice should be encouraged to continue until each pilot knows how to manage all of the parameters in all of their combinations. It is not known to the authors of this report what the Vietnam pilots know and believe about the real limits of high-speed recoveries and how they developed their knowledge, skill, and feelings. The results of querying one Vietnam pilot are included in this discussion. This is a very small sample, and the querying method was not at all sophisticated; yet the perspective and insight gained are very valuable. It is beyond the scope of this study to develop a sophisticated questionnaire for a larger sample of pilots who flew the AH-1G in Vietnam; however, this is a task which should be done as quickly as possible before these pilots are dispersed or their recollections dimmed.

Such information from a larger sample could add an important dimension to the probing for the real limits and desirable characteristics of new-generation armed helicopters. The results of this larger poll would answer the question at hand concerning the maximum dive speed and the low frequency of occurrence of speeds in the 160- to 190-knot range reported in Reference 8. Was the frequency of occurrence due to (a) the fact that it was not a useful tactic, (b) real or imagined physical abuse of the helicopter, (c) degradation of the gun platform, or (d) physical discomfort, unmanageable workload, or apprehension during dive recovery? Factors affecting the smartness of dive recovery are enumerated in the following paragraphs.

The following phenomena tend to limit a symmetrical pullup:

(a) Overspeeding the rotor. Application of aft cyclic tends to put the rotor in autorotation, cones the rotor, and washes out collective pitch via pitch-cone coupling. Both autorotation and pitch reduction tend to overspeed the rotor; thus cyclic and collective rate and magnitude must be coordinated to limit rotor speed.

(b) Rapid buildup in pitch rate and attitude causing loss of visual reference, which is the primary cue for exiting the maneuver via a pushover or pedal turn. If the pullup is held too long, the only coordinated maneuver for recovery is a loop. This has reportedly been accomplished, but it is not recommended since failure to follow through can have disastrous consequences. Uncoordinated exits are not recommended because not enough is known about them; especially worrisome is the potential for exceeding the flapping clearance and striking the flapping stops.

(c) Unloading the rotor. The pushover which may follow the pullup tends to unload the rotor and decrease control power. Flight at or below 0g is prohibited since, with SCAS off, a divergent right roll can develop which cannot be controlled by the instinctive left cyclic reaction.

(d) Mushing of the rotor. This is caused by stall or pitch-cone coupling, or both, as discussed in the Operator's Manual, Reference 13. The Vietnam pilot reported this (and the accompanying loss of altitude) as the primary deterrent to a severe pullup. Bell pilots maintain that mushing can be mitigated by relying more heavily on the cyclic flare for building the initial normal load factor.

A coordinated application of cyclic and collective is then used to control rpm and build additional load factor.

(e) Boost feedback. Cyclic or collective feedback, or both, can occur in high-g maneuvers. The symmetrical pullups of the structural demonstration were limited by collective feedback, while cyclic feedback limited similar maneuvers in Reference 14.

Both cyclic and collective feedback are due to high oscillatory loads with a rational pattern which is explainable but not yet predictable by analysis. There is some controversy as to whether the conditions for occurrence are repeatable. The harmonic content of the loads and the transfer pattern from the rotating to the fixed control system are rationally related to the blade frequency diagram and the transfer trigonometry. The airloads are less understood but are most likely a growth in regular oscillatory loads plus impulsive stall loads and unsteady airloads from vorticity. The rotor modes involved are the first inplane and the first and second out-of-plane asymmetrical modes. Torsional modes are not involved; thus there is no reason to think of this as stall flutter.

There is an abundance of flight test data for constructing a clear and rational picture of the sequence of events during feedback for current design guidance and future analytical guidance; however, this has not been deemed necessary to date for the following reason. During development flight testing, the boost capability is increased to a point where boost feedback does not occur so early as to seriously restrict the flight envelope nor so late as to allow an unannounced buildup of dangerous rotor oscillatory loads. The pilots have a very strong feeling about this iteration. They rely on boost feedback as the primary warning that the rotor is being abused, and it is their signal to back off. An increase in vibration usually accompanies the load buildup on current helicopters, but this will be less true in the future since more effective isolation systems are coming into use. Because of this and the fact that the vibration in the stick commands more attention than vibration in the structure, pilots do not want to do away with boost feedback.

The symmetrical pullup followed by a pedal turn is reported to be easier to execute and less taxing on the helicopter than the pullup and pushover. An important factor is that the pilot does not lose visual reference. The rolling pullup also presents the pilot with an easier work load than the symmetrical pullup and pushover. Loss of visual reference, overspeed, mushing, and 0g are mitigated or absent. The limiting phenomena will likely be boost feedback, vibration, or overtorquing. In the case of a left roll, a 10- to 15-psi torque surge can occur.

From the foregoing, it is concluded that the maximum speed of 185 knots at 7546 pounds was probably limited by vibration in the dive and vibration and boost feedback in the recovery. However, the infrequent use of the high-speed level-flight capability of the AH-1G must be explained in terms of local requirements.

- Low requirement for long, high-speed escort missions
- Infrequent need for long, high-speed dashes from base to scene of action
- Difficulty of locating and attacking mobile enemy troops in protective ground cover when flying at high speed
- Reduced visibility and sighting time because of required pullout height and rapid closure rate
- Absence of concentrated, sophisticated antiaircraft installations, etc.

ROTOR RPM

The rotor rpm distributions show that more than 95 percent of the time in both data samples was flown between 310 and 325 rpm with the normal steady value being between 315 and 318. It appears that rotor rpm was controlled within limits, with very few points falling above 330 rpm. A very small amount of time was recorded in the 340- to 355-rpm range during the maneuver segment. All such recording occurred during descents. The highest value of 351 rpm was recorded during a descent which appeared to approach an autorotation.

During design and development testing, a great deal of attention is paid to rotor rpm since it optimizes and sizes so many details of the helicopter: rotor and engine performance; shafting and bearings; rotor strength and fatigue life; natural frequencies of the rotor, pylon, fuselage, and driveshafts; rotor noise; etc. The final choice of rpm range is frequently a compromise made during development testing. The band adopted for the AH-1G and listed in the Operator's Manual is 294 to 324 rpm. Not all rpm's in this band are equally desirable; 324 is considered the normal, or preferred, rpm. It is interesting to note that the data peak was in the 315- to 318-rpm band. The Vietnam pilot queried said it was commonly believed that operating at less than design maximum was less abusive of the helicopter. The AH-1G is not overly sensitive to this rpm spread; however, this is not always the case. Often rotor loads and vibration are quite sensitive to such a spread, being worse at lower rpm. Possibly the Operator's Manual should state the preferred rpm and list the reasons. More information about why 315 to 318 rpm was used plus a finer breakout of rpm would be a useful contribution to the study.

The good rpm control shown by the data suggests that it was a simple task, while several other sources suggest that rpm control is difficult. Consider the following excerpt from Reference 20.

> "The maximum angle-of-attack capability of the AH-1G during pullups is severely restricted by a rapid buildup of rotor rpm. Pilot workload to control rpm is frequently excessive. The power-off upper rotor speed limit (339 rpm) is marked with a red line on the aircraft instruments and is interpreted by the pilot as a not-to-exceed rotor speed. The 339-rpm limit allows less than a 5-percent overspeed from the normal operating value of 324. This small margin is considered a design shortcoming. The military specification for structural design requirements for helicopters requires a 25-percent margin between design maximum and power-on limit rotor speed. A minimum margin of 10-percent between design maximum and limit operating rpm should be specified for Army helicopters."

It is important to know if the pilots found it easy or difficult to control rpm. Did they bias the data? Again, a wider poll of Vietnam pilots would help to answer this question. If the control evidenced by the data was relatively easy, then the 5-percent overspeed margin used for the AH-1G is validated by the data.

Concerning the reason for the maximum one-time occurrence and the neighboring values, they all occurred during descents. This is the most likely condition for such overspeeds, especially if the descent started at, say, 5000 feet and terminated in a cyclic pullup, with primary attention of the pilot being on the target.

## NORMAL ACCELERATION

A maximum normal acceleration of 2.4g's was measured several times in both samples of data. This maximum was most probably encountered in a turn. At rotor thrusts of 16,500 to 18,000 pounds, control feedback forces are encountered which provide a warning to the pilot. The BHC Model 540 rotor hub and control geometry also provides a pitch-cone coupling which tends to reduce blade pitch when the rotor cones upward. This has a limiting influence on the load factor which is developed in maneuvering flight. The extreme minimum normal acceleration of 0.1g probably occurred during a pushover. Since there is no g-meter in the helicopter, the pilot must rely on instinct. The helicopter pilot's tolerance to g levels in this range is generally quite low. This tolerance, plus an awareness of the warning in the Operator's Manual, could provide cause for terminating this maneuver.

## TORQUE

The torque distributions for the two samples are rational in themselves and compared to each other. The aircraft were probably taking off fully loaded and flying at about the same height above the ground. The higher temperatures of the Sample II data provided high density altitudes, requiring higher power and higher torques than Sample I.

The ratio of the percentage of time above 80°F of Sample II to Sample I was 1.25, and the ratio of percentage of time above 40 psi was 1.35. There was not much chance for overtorquing since a relatively small percentage of time was spent above 40 psi - 12.6 percent for Sample I and 16.1 percent for Sample II. The data indicate that the aircraft were operating within the constraint of available power, which was generally below the torque limit. If greater power had been available, overtorquing might have been more frequent.

Some valuable information which could influence the rationale for the transmission restriction could be derived from the two data samples. This would require a finer breakdown and clearer explanation in terms of the oscillograph trace, and it would be instructive to compare the finer histograms with the oscillograph traces themselves. The limited number of trace samples contained in Reference 17 show that the torque pressure has a complex wave form reflecting the varying torque demand experienced during maneuvering under combat conditions.

The data presented do not produce a strong physical picture of events that were going on due to transient excitation of the first torsional mode, energy maneuverability, power increase in dives (Reference 20), left rolls (Reference 19), etc. Further, it would be useful to know how well the pilot's torque meter followed the oscillograph galvanometer.

The maximum engine torque of greater than 60 psi occurred under conditions of 0 to 1000 feet altitude, 0 to 40 knots airspeed, 8000 to 9000 pounds gross weight, 0 to 300 feet-per-minute rate of climb, 310 to 325 rotor rpm, and an OAT of 60° to 70°F. The low altitude and airspeed indicate that the overtorque occurred during takeoff. The most probable event was the requirement to take off over an obstacle under conditions requiring high power. The extremes of the above parameter ranges - 1000 feet, 0 knots, 9000 pounds, 300 feet per minute, 310 rpm, 60°F - would be more than sufficient as a condition.

## GROSS WEIGHT

The maximum recorded gross weight of 9522 pounds indicates that the pilots were paying close attention to the operating limit of 9500 pounds. This was encouraged by requiring the pilots to fill out supplemental information sheets indicating the fuel and armament loading at takeoff. It is probable that, had this not been required, the helicopters would have been flown more frequently at the maximum overload weight at which they could get airborne. The Vietnam pilot and others who observed Vietnam helicopter operations report that the AH-1G's frequently took off with the maximum liftable weight. Because of the temperature's effect on power, this would constitute an overload only in the early morning; thus, there was little opportunity to seriously overload the AH-1G during the Vietnam operation. However, this could be a problem in a cooler climate.

Under combat conditions, taking off with the maximum liftable load would be a very natural thing to do, and whether this is right or not would have to be judged for each case. For example, a field commander will probably not concern himself with meeting established life and overhaul schedules if the intensity of battle is such that the probable survival life is much shorter than the overhaul life. Under these conditions, resourceful pilots will experiment with overload configurations which meet an immediate need. If no obvious limitation or problem results, the configuration will be adopted. This is not necessarily bad, and it may or may not lead to a problem later. However, such practices should be reported to cognizant Army technical agencies and the contractor for evaluation and information. This situation should be provided for in the maintainability/reliability program and the design/development loop for future helicopters; for existing helicopters, the safety of flight release via the contractor should be more conscientiously applied. The foregoing again points out the need for a wider poll of Vietnam veterans to assess bias of the data.

## CONSIDERATIONS FOR CHANGES IN STRUCTURAL DESIGN CRITERIA

Based only on the results of the values of maximum one-time occurrences measured and reported in Reference 8, no changes in the structural design criteria used for the AH-1G can be recommended. Only the gross weight was shown to have exceeded the value specified in the structural design criteria, and this was by an insignificant amount.

Although the 1.05 factor was shown to adequately cover rotor overspeed conditions for the 540 rotor system, the advisability of recommending a change in the structural design criteria to reduce the design limit rotor speed factor from 1.25 to some lesser value could be questioned. As mentioned earlier, one of the initial design objectives in the 540 rotor system was to develop a rotor with high rotational inertia to provide increased safety in emergency autorotational landings. A higher inertia system accelerates less in a cyclic flare and gives the pilot more time to respond and apply corrective control in case of rotor overspeed. This suggests that some flexibility in specifying rotor overspeed factors could be incorporated in future design criteria to account for differences in rotational inertia. For the 540 rotor system, a value of 1.10 would appear to be a realistic factor which still contained a reasonable degree of conservatism.

A study comparing maximum measured rotor speeds from operational surveys of other types and models of helicopters would provide information to establish the validity of some rotor inertia/rotor overspeed relationship.

## PREDICTION OF LIMITING PARAMETERS

Because of the large number of parameters and the complex coupling and interactions involved, it would appear that there is no simple method for predicting maximum one-time occurrences. The development of a comprehensive mathematical model capable of handling all of the known significant parameters with provisions for "flying in the computer" would appear to be the best approach. With this tool, the vehicle limits could be explored and maximum values could be determined for specific parameters.

In addition to the limits established for the vehicle, pilot-imposed limits should be considered. The interface of vehicle and pilot could be handled in several different ways. While fixed- or moving-base simulators could be used for pilot input and response, this degree of sophistication might not be required. Since human factors information is available in the form of tolerance to vibration level, pitch rates, control input strength, attitude, etc., these limits could be super-imposed on plots of vehicle limits to determine the probable maximum values for the parameters of interest. The super-imposed plots of vehicle and pilot limits would then indicate areas where auxiliary systems such as warning devices or control rate limiters should be provided to ensure safe operation of the helicopter.

## CONCLUSIONS

Based on the results of this study, it is concluded that:

1. Comparison of the original engineering frequency-of-occurrence spectrum and the Southeast Asia mission profile data showed that the AH-1G was being operated at higher gross weights and lower airspeeds for larger percentages of time than originally estimated.
2. These differences in use had compensating effects which caused the fatigue lives calculated by the original engineering spectrum and the Southeast Asia spectrum to be nearly the same.
3. The attack helicopter spectrum shown in Navy AR-56 is very similar to the original AH-1G engineering frequency-of-occurrence spectrum.
4. Of the maximum one-time values of load-sensitive parameters measured in the Southeast Asia operation survey, only the gross weight was shown to have exceeded the value specified in the structural design criteria, and this was by an insignificant amount.

## RECOMMENDATIONS

The following recommendations are based upon the study presented in this report.

1. In presenting data from future operational surveys, an effort should be made to establish histograms with smaller class intervals.

2. The presentation of maximum one-time occurrence data should be in the form of a short-time history of all parameters measured during the event.

3. Consideration should be given to reducing the factor used to establish design limit rotor speed from 1.25 to 1.10 for high-inertia rotor systems.

4. Work in the area of developing analytical methods for predicting helicopter operating limits and loads associated with these limits should be encouraged.

5. A questionnaire regarding vehicle use and limits should be developed to quiz pilots who flew the AH-1G in Vietnam.

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APPENDIX
FATIGUE LIFE DETERMINATION

TABLE VIII. 540-011-250-1 MAIN ROTOR BLADE FATIGUE LIFE DETERMINATION

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| I.GROUND CONDITIONS | | | | | |
| A.NORMAL START | 0.4000 | 7656 | 0 FA | | 0.0 |
| B.SHUTDOWN W/COLL. | 0.4000 | 7656 | 0 FA | | 0.0 |
| II.IGE MANEUVERS | | | | | |
| A.TAKE-OFF | | | | | |
| 1.NORMAL | 0.0511 | 978 | 529 AA | | 0.0 |
| | 0.7668 | 14677 | 363 BA | | 0.0 |
| | 0.4601 | 8806 | 324 CA | | 0.0 |
| 2.JUMP | 0.0057 | 109 | 299 AA | | 0.0 |
| | 0.0852 | 1631 | 391 BA | | 0.0 |
| | 0.0511 | 978 | 397 CA | | 0.0 |
| B.HOVERING | | | | | |
| 1.STEADY | 0.0800 | 1531 | 228 AA | | 0.0 |
| | 1.2000 | 22968 | 310 BA | | 0.0 |
| | 0.7200 | 13781 | 340 CA | | 0.0 |
| 2.RIGHT TURN | 0.0067 | 128 | 380 AA | | 0.0 |
| | 0.1002 | 1918 | 284 BA | | 0.0 |
| | 0.0601 | 1151 | 354 CA | | 0.0 |
| 3.LEFT TURN | 0.0067 | 128 | 337 AA | | 0.0 |
| | 0.1002 | 1918 | 376 BA | | 0.0 |
| | 0.0601 | 1151 | 322 CA | | 0.0 |
| 4.CONTROL CORR. | | | | | |
| (A).LONGITUDINAL | 0.0007 | 13 | 1364 AA | 2.365 | 0.000005 |
| | 0.0100 | 192 | 1118 BA | 13.868 | 0.000014 |
| | 0.0060 | 115 | 1073 CA | 22.058 | 0.000005 |
| (B).LATERAL | 0.0007 | 13 | 939 AA | | 0.0 |
| | 0.0100 | 192 | 959 BA | | 0.0 |
| | 0.0060 | 115 | 834 CA | | 0.0 |
| (C).RUDDER | 0.0007 | 13 | 0 AC | | 0.0 |
| | 0.0100 | 192 | 506 BA | | 0.0 |
| | 0.0060 | 115 | 491 CA | | 0.0 |
| C.SIDEWARD FLIGHT | | | | | |
| 1.TO THE RIGHT | 0.0096 | 184 | 487 AA | | 0.0 |
| | 0.1442 | 2761 | 591 BA | | 0.0 |
| | 0.0865 | 1656 | 636 CA | | 0.0 |
| 2.TO THE LEFT | 0.0096 | 184 | 312 AA | | 0.0 |
| | 0.1442 | 2761 | 402 BA | | 0.0 |
| | 0.0865 | 1656 | 471 CA | | 0.0 |
| D.REARWARD FLIGHT | 0.0096 | 184 | 493 AA | | 0.0 |
| | 0.1442 | 2761 | 650 BA | | 0.0 |
| | 0.0865 | 1656 | 875 CA | | 0.0 |
| E.ACCELERATION | | | | | |
| HOVER TO CLIMB A/S | 0.0200 | 383 | 578 AA | | 0.0 |
| | 0.3000 | 5742 | 565 BA | | 0.0 |
| | 0.1800 | 3445 | 652 CA | | 0.0 |

TABLE VIII (Continued)

| FLIGHT CONDITION | | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|---|
| F.DECELERATION | | | | | | |
| 1.NORMAL | | 0.0200 | 383 | 695 AA | | 0.0 |
| | | 0.3000 | 5742 | 848 BA | | 0.0 |
| | | 0.1800 | 3445 | 900 CA | | 0.0 |
| 2.QUICK STOP | | 0.0040 | 77 | 818 AA | | 0.0 |
| | | 0.0600 | 1148 | 1015 BA | 44.531 | 0.000026 |
| | | 0.0360 | 689 | 1147 CA | 10.686 | 0.000064 |
| G.APPR. AND LANDING | | 0.2204 | 4218 | 705 AA | | 0.0 |
| | | 3.3057 | 63271 | 1070 BA | 22.708 | 0.002786 |
| | | 1.9834 | 37963 | 1051 CA | 28.202 | 0.001346 |
| III.FORWARD LEVEL FLIGHT | | | | | | |
| AIRSPEED | RPM | | | | | |
| A. 0.50 VH | 314 | 0.0104 | 196 | 387 AA | | 0.0 |
| | | 0.1563 | 2945 | 403 BA | | 0.0 |
| | | 0.0938 | 1767 | 512 CA | | 0.0 |
| | 324 | 0.0938 | 1823 | 327 AA | | 0.0 |
| | | 1.4070 | 27352 | 442 BA | | 0.0 |
| | | 0.8442 | 16411 | 444 CA | | 0.0 |
| B. 0.60 VH | 314 | 0.0309 | 582 | 497 AA | | 0.0 |
| | | 0.4634 | 8730 | 583 BA | | 0.0 |
| | | 0.2780 | 5238 | 557 CA | | 0.0 |
| | 324 | 0.2780 | 5405 | 468 AA | | 0.0 |
| | | 4.1705 | 81074 | 483 BA | | 0.0 |
| | | 2.5023 | 48644 | 580 CA | | 0.0 |
| C. 0.70 VH | 314 | 0.0342 | 644 | 501 AA | | 0.0 |
| | | 0.5131 | 9666 | 614 BA | | 0.0 |
| | | 0.3078 | 5800 | 769 CA | | 0.0 |
| | 324 | 0.3079 | 5985 | 512 AA | | 0.0 |
| | | 4.6178 | 89770 | 598 BA | | 0.0 |
| | | 2.7707 | 53862 | 662 CA | | 0.0 |
| D. 0.80 VH | 314 | 0.0551 | 1038 | 639 AA | | 0.0 |
| | | 0.8264 | 15569 | 628 BA | | 0.0 |
| | | 0.4958 | 9341 | 757 CA | | 0.0 |
| | 324 | 0.4958 | 9639 | 584 AA | | 0.0 |
| | | 7.4377 | 144588 | 553 BA | | 0.0 |
| | | 4.4626 | 86753 | 707 CA | | 0.0 |
| E. 0.90 VH | 314 | 0.0160 | 301 | 944 AA | | 0.0 |
| | | 0.2394 | 4510 | 728 BA | | 0.0 |
| | | 0.1436 | 2706 | 758 CA | | 0.0 |
| | 324 | 0.1436 | 2792 | 848 AA | | 0.0 |
| | | 2.1546 | 41885 | 634 BA | | 0.0 |
| | | 1.2928 | 25131 | 675 CA | | 0.0 |
| F. VH | 314 | 0.0138 | 261 | 1261 AA | 4.440 | 0.000059 |
| | | 0.2076 | 3911 | 989 BA | 65.076 | 0.000060 |
| | | 0.1246 | 2347 | 988 CA | 65.699 | 0.000036 |

TABLE VIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| 324 | 0.1246 | 2421 | 1029 AA | 36.977 | 0.000065 |
| | 1.8684 | 36322 | 961 BA | | 0.0 |
| | 1.1210 | 21793 | 908 CA | | 0.0 |
| IV.NON-FIRING MANEUVERS | | | | | |
| A.FULL POWER CLIMB | | | | | |
| 1.NORMAL | 0.1000 | 1914 | 428 AA | | 0.0 |
| | 1.5000 | 28710 | 545 BA | | 0.0 |
| | 0.9000 | 17226 | 465 CA | | 0.0 |
| 2.HIGH-SPEED | 0.0017 | 33 | 695 AA | | 0.0 |
| | 0.0256 | 489 | 890 BA | | 0.0 |
| | 0.0153 | 294 | 1098 CA | 16.862 | 0.000017 |
| B.MAXIMUM RATE ACCEL | | | | | |
| CLIMB - CRUISE A/S | 0.1870 | 3580 | 865 AA | | 0.0 |
| | 2.8056 | 53699 | 753 BA | | 0.0 |
| | 1.6834 | 32219 | 817 CA | | 0.0 |
| C.NORMAL TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0668 | 1279 | 574 AA | | 0.0 |
| | 1.0020 | 19178 | 672 BA | | 0.0 |
| | 0.6012 | 11507 | 668 CA | | 0.0 |
| (B) 0.7 VH | 0.0668 | 1279 | 674 AA | | 0.0 |
| | 1.0020 | 19178 | 968 BA | 90.757 | 0.000211 |
| | 0.6012 | 11507 | 897 CA | | 0.0 |
| (C) 0.9 VH | 0.0043 | 83 | 823 AA | | 0.0 |
| | 0.0652 | 1247 | 903 BA | | 0.0 |
| | 0.0391 | 748 | 1142 CA | * | 0.000007 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0668 | 1279 | 564 AA | | 0.0 |
| | 1.0020 | 19178 | 592 BA | | 0.0 |
| | 0.6012 | 11507 | 695 CA | | 0.0 |
| (B) 0.7 VH | 0.0668 | 1279 | 694 AA | | 0.0 |
| | 1.0020 | 19178 | 891 BA | | 0.0 |
| | 0.6012 | 11507 | 907 CA | | 0.0 |
| (C) 0.9 VH | 0.0043 | 83 | 796 AA | | 0.0 |
| | 0.0652 | 1247 | 856 BA | | 0.0 |
| | 0.0391 | 748 | 940 CA | | 0.0 |
| D. .9 VH CONTR. CORR | | | | | |
| 1.LONGITUDINAL | 0.0033 | 64 | 1364 AA | 2.361 | 0.000027 |
| | 0.0501 | 959 | 1429 BA | 1.676 | 0.000572 |
| | 0.0301 | 575 | 1046 CA | 30.052 | 0.000019 |
| 2.LATERAL | 0.0033 | 64 | 1300 AA | 3.437 | 0.000019 |
| | 0.0501 | 959 | 1246 BA | 4.914 | 0.000195 |
| | 0.0301 | 575 | 1308 CA | 3.272 | 0.000176 |
| 3.RUDDER | 0.0006 | 11 | 953 AA | | 0.0 |
| | 0.0085 | 162 | 916 BA | | 0.0 |

TABLE VIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0051 | 97 | 841 CA | | 0.0 |
| E.SIDESLIP | 0.0080 | 153 | 382 AA | | 0.0 |
| | 0.1200 | 2297 | 391 BA | | 0.0 |
| | 0.0720 | 1378 | 588 CA | | 0.0 |
| F.PART POWER DESCENT | 0.0040 | 77 | 637 AA | | 0.0 |
| | 0.0600 | 1148 | 548 BA | | 0.0 |
| | 0.0360 | 689 | 679 CA | | 0.0 |
| V.GUNNERY MANEUVERS | | | | | |
| A.FIRING IN A HOVER | 0.0050 | 96 | 0 AC | | 0.0 |
| | 0.0751 | 1438 | 0 BC | | 0.0 |
| | 0.0451 | 863 | 0 CC | | 0.0 |
| B.STRAFING IN ACCEL. FROM A HOVER | 0.0033 | 64 | 0 AC | | 0.0 |
| | 0.0501 | 959 | 0 BC | | 0.0 |
| | 0.0301 | 575 | 0 CC | | 0.0 |
| C.GUNNERY RUNS | | | | | |
| 1.PT. TARGET DIVES | | | | | |
| (A) TO 0.6 VL | 0.0187 | 358 | 324 AA | | 0.0 |
| | 0.2806 | 5370 | 393 BA | | 0.0 |
| | 0.1683 | 3222 | 679 CA | | 0.0 |
| (B) TO 0.8 VL | 0.1040 | 1991 | 619 AA | | 0.0 |
| | 1.5602 | 29862 | 861 BA | | 0.0 |
| | 0.9361 | 17917 | 989 CA | 64.876 | 0.000276 |
| (C) TO 0.9 VL | 0.3220 | 6164 | 852 AA | | 0.0 |
| | 4.8305 | 92455 | 1106 BA | * | 0.000559 |
| | 2.8983 | 55473 | 1240 CA | * | 0.000938 |
| (D) TO VL | 0.0008 | 15 | 1267 AA | 4.272 | 0.000004 |
| | 0.0120 | 230 | 1404 BA | * | 0.000002 |
| | 0.0072 | 138 | 1598 CA | * | 0.000031 |
| 2.SPRAY FIRE DIVES | | | | | |
| (A) TO 0.6 VL | 0.0080 | 153 | 306 AA | | 0.0 |
| | 0.1202 | 2301 | 443 BA | | 0.0 |
| | 0.0721 | 1381 | 690 CA | | 0.0 |
| (B) TO 0.8 VL | 0.1079 | 2065 | 510 AA | | 0.0 |
| | 1.6184 | 30977 | 936 BA | | 0.0 |
| | 0.9711 | 18586 | 1030 CA | 36.782 | 0.000505 |
| (C) TO 0.9 VL | 0.2291 | 4384 | 988 AA | 65.556 | 0.000067 |
| | 3.4358 | 65762 | 1300 BA | * | 0.000928 |
| | 2.0615 | 39457 | 1337 CA | * | 0.002256 |
| (D) TO VL | 0.0040 | 77 | 1036 AA | 33.905 | 0.000002 |
| | 0.0600 | 1148 | 846 BA | | 0.0 |
| | 0.0360 | 689 | 1754 CA | * | 0.000244 |
| D.GUNNERY RUN P/U | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.6 VL | 0.0020 | 38 | 751 AA | | 0.0 |

TABLE VIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0300 | 574 | 839 BA | | 0.0 |
| | 0.0180 | 345 | 1199 CA | 6.951 | 0.000050 |
| (B) 0.8 VL | 0.0040 | 77 | 1204 AA | 6.660 | 0.000011 |
| | 0.0600 | 1148 | 1251 BA | 4.758 | 0.000241 |
| | 0.0360 | 689 | 1460 CA | * | 0.000168 |
| (C) 0.9 VL | 0.0100 | 191 | 1449 AA | * | 0.000013 |
| | 0.1500 | 2871 | 1542 BA | * | 0.000886 |
| | 0.0900 | 1723 | 1676 CA | * | 0.001068 |
| (D) VL | 0.0067 | 128 | 1804 AA | * | 0.000139 |
| | 0.1002 | 1918 | 1925 BA | * | 0.001643 |
| | 0.0601 | 1151 | 1944 CA | * | 0.002635 |
| 2.TO THE LEFT | | | | | |
| (A) 0.6 VL | 0.0020 | 38 | 649 AA | | 0.0 |
| | 0.0300 | 574 | 1067 BA | 23.489 | 0.000024 |
| | 0.0180 | 345 | 1239 CA | 5.177 | 0.000067 |
| (B) 0.8 VL | 0.0040 | 77 | 1131 AA | 12.364 | 0.000006 |
| | 0.0600 | 1148 | 1257 BA | 4.566 | 0.000251 |
| | 0.0360 | 689 | 1579 CA | * | 0.000200 |
| (C) 0.9 VL | 0.0100 | 191 | 1364 AA | * | 0.000014 |
| | 0.1500 | 2871 | 1517 BA | * | 0.000689 |
| | 0.0900 | 1723 | 1745 CA | * | 0.001546 |
| (D) VL | 0.0067 | 128 | 1724 AA | 0.491 | 0.000260 |
| | 0.1002 | 1918 | 1865 BA | * | 0.001617 |
| | 0.0601 | 1151 | 2234 CA | * | 0.003525 |
| 3.SYMMETRICAL | | | | | |
| (A) 0.6 VL | 0.0002 | 4 | 871 AA | | 0.0 |
| | 0.0030 | 57 | 1071 BA | 22.390 | 0.000003 |
| | 0.0018 | 34 | 1239 CA | 5.175 | 0.000007 |
| (B) 0.8 VL | 0.0020 | 38 | 1045 AA | 30.230 | 0.000001 |
| | 0.0301 | 575 | 1302 BA | 3.413 | 0.000169 |
| | 0.0180 | 345 | 1446 CA | 1.535 | 0.000225 |
| (C) 0.9 VL | 0.0033 | 64 | 1451 AA | 1.501 | 0.000043 |
| | 0.0501 | 959 | 1598 BA | 0.783 | 0.001225 |
| | 0.0301 | 575 | 1779 CA | 0.407 | 0.001413 |
| (D) VL | 0.0007 | 13 | 2071 AA | 0.175 | 0.000073 |
| | 0.0100 | 192 | 1943 BA | 0.247 | 0.000778 |
| | 0.0060 | 115 | 2102 CA | 0.162 | 0.000711 |
| E.GUNNERY TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0250 | 479 | 803 AA | | 0.0 |
| | 0.3757 | 7191 | 1000 BA | 55.081 | 0.000131 |
| | 0.2254 | 4315 | 1158 CA | 9.679 | 0.000446 |
| (B) 0.7 VH | 0.0729 | 1396 | 1031 AA | 36.001 | 0.000039 |
| | 1.0942 | 20942 | 1308 BA | * | 0.000753 |
| | 0.6565 | 12565 | 1148 CA | 10.586 | 0.001187 |
| (C) 0.9 VH | 0.0016 | 31 | 1870 AA | * | 0.000007 |

TABLE VIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0240 | 459 | 1841 BA | * | 0.000178 |
| | 0.0144 | 276 | 1766 CA | * | 0.000094 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0250 | 479 | 930 AA | | 0.0 |
| | 0.3757 | 7191 | 918 BA | | 0.0 |
| | 0.2254 | 4315 | 1252 CA | 4.7[illegible]9 | 0.000915 |
| (B) 0.7 VH | 0.0729 | 1396 | 952 AA | | 0.0 |
| | 1.0942 | 20942 | 1049 BA | 28.864 | 0.000726 |
| | 0.6565 | 12565 | 1174 CA | 8.456 | 0.001486 |
| (C) 0.9 VH | 0.0016 | 31 | 1141 AA | 11.226 | 0.000003 |
| | 0.0240 | 459 | 1339 BA | * | 0.000014 |
| | 0.0144 | 276 | 1578 CA | * | 0.000063 |
| F. S-TURNS | | | | | |
| 1.AT 0.8 VH | 0.0040 | 77 | 1205 AA | 6.623 | 0.000012 |
| | 0.0600 | 1148 | 1847 BA | * | 0.000306 |
| | 0.0360 | 689 | 2108 CA | * | 0.000404 |
| 2.AT VH | 0.0051 | 98 | 1522 AA | 1.080 | 0.000091 |
| | 0.0769 | 1472 | 1849 BA | * | 0.000773 |
| | 0.0462 | 883 | 2034 CA | * | 0.000561 |
| VI.POWER TRANSITIONS | | | | | |
| A.POWER TO AUTO | | | | | |
| 1. 0.5 VH | 0.0033 | 64 | 383 AA | | 0.0 |
| | 0.0501 | 959 | 453 BA | | 0.0 |
| | 0.0301 | 575 | 529 CA | | 0.0 |
| 2. 0.7 VH | 0.0083 | 160 | 463 AA | | 0.0 |
| | 0.1252 | 2397 | 603 BA | | 0.0 |
| | 0.0751 | 1438 | 770 CA | | 0.0 |
| 3. 0.9 VH | 0.0060 | 115 | 809 AA | | 0.0 |
| | 0.0900 | 1723 | 742 BA | | 0.0 |
| | 0.0540 | 1034 | 765 CA | | 0.0 |
| B.AUTO TO POWER | | | | | |
| 1.IN GROUND-EFFECT | 0.0040 | 77 | 830 AA | | 0.0 |
| | 0.0600 | 1148 | 809 BA | | 0.0 |
| | 0.0360 | 689 | 1188 CA | 7.596 | 0.000091 |
| 2. 0.4 VH | 0.0067 | 128 | 541 AA | | 0.0 |
| | 0.1002 | 1918 | 560 BA | | 0.0 |
| | 0.0601 | 1151 | 824 CA | | 0.0 |
| 3. 0.6 VH | 0.0324 | 620 | 902 AA | | 0.0 |
| | 0.4862 | 9307 | 924 BA | | 0.0 |
| | 0.2917 | 5584 | 1080 CA | 20.320 | 0.000275 |
| 4. MAX AUTO A/S | 0.0012 | 23 | 1008 AA | 49.085 | 0.000000 |
| | 0.0180 | 345 | 964 BA | 97.315 | 0.000004 |
| | 0.0108 | 207 | 1375 CA | 2.224 | 0.000093 |
| VII.AUTOROTATION | | | | | |

TABLE VIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY M/R BLADE SPAR STRESS STA. 41.0 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| A.STABILIZED FLIGHT | | | | | |
| 1. 0.4 VH | 0.0043 | 83 | 322 AA | | 0.0 |
| | 0.0652 | 1247 | 330 BA | | 0.0 |
| | 0.0391 | 748 | 391 CA | | 0.0 |
| 2. 0.6 VH | 0.0809 | 1549 | 393 AA | | 0.0 |
| | 1.2142 | 23239 | 419 BA | | 0.0 |
| | 0.7285 | 13943 | 435 CA | | 0.0 |
| 3. MAX AUTO A/S | 0.0040 | 77 | 553 AA | | 0.0 |
| | 0.0600 | 1148 | 624 BA | | 0.0 |
| | 0.0360 | 689 | 622 CA | | 0.0 |
| B.AUTO TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.4 VH | 0.0033 | 64 | 526 AA | | 0.0 |
| | 0.0501 | 959 | 477 BA | | 0.0 |
| | 0.0301 | 575 | 484 CA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 1036 | 627 AA | | 0.0 |
| | 0.8119 | 15540 | 604 BA | | 0.0 |
| | 0.4872 | 9324 | 553 CA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 38 | 778 AA | | 0.0 |
| | 0.0300 | 574 | 822 BA | | 0.0 |
| | 0.0180 | 345 | 625 CA | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.4 VH | 0.0033 | 64 | 429 AA | | 0.0 |
| | 0.0501 | 959 | 447 BA | | 0.0 |
| | 0.0301 | 575 | 434 CA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 1036 | 593 AA | | 0.0 |
| | 0.8119 | 15540 | 697 BA | | 0.0 |
| | 0.4872 | 9324 | 520 CA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 38 | 777 AA | | 0.0 |
| | 0.0300 | 574 | 789 BA | | 0.0 |
| | 0.0180 | 345 | 703 CA | | 0.0 |
| C.AUTO LANDING | 0.0040 | 77 | 1424 AA | 1.718 | 0.000045 |
| | 0.0600 | 1148 | 1293 BA | 3.599 | 0.000319 |
| | 0.0360 | 689 | 2189 CA | * | 0.000142 |

ENDURANCE LIMIT = 962.0 TOTAL DAMAGE (D) = 0.040377
MATERIAL = ALUM
FREQUENCY = 1 / REV OF M/R FATIGUE LIFE = 100/D = 2476 HOURS
* DAMAGE CALCULATED FROM MEASURED LOAD FREQUENCIES.

## TABLE IX. 540-011-154-5 MAIN ROTOR GRIP FATIGUE LIFE DETERMINATION

| Flight Condition | Frequency of Occurrence % of Total Flt Time | Cycles in 100 Hours (n) | Oscillatory Stress in Lower Grip Tang--PSI | Cycles to Failure (N)x$10^{-6}$ | Damage Fraction (n/N) |
|---|---|---|---|---|---|
| $V_H$ Level Flt | | | | | |
| 314 RPM | 0.0138 | | 4484 AB | | |
| | 0.2076 | | 2981 DA | | |
| | 0.1246 | | 3521 FA | | |
| 324 RPM | 0.1246 | | 3785 AB | | |
| | 1.8684 | | 3355 DA | | |
| | 1.1210 | | 3334 FA | | |
| S-Turn | | | | | |
| 0.8 $V_H$ | 0.0040 | | 4903 AA | | |
| | 0.0600 | | 5497 CA | | |
| | 0.0360 | | 5604 FB | | |
| $V_H$ | 0.0051 | | 6202 AA | | |
| | 0.0769 | | 5485 CA | | |
| | 0.0462 | 883 | 7037 FA | 0.40 | 0.002200 |

Total Damage in 100 Hr. (D) = 0.002200

Fatigue Life = 100/D = 45,454 Hours

TABLE X. 540-011-153-13 MAIN ROTOR YOKE EXTENSION FATIGUE LIFE DETERMINATION

| Flight Condition | Frequency of Occurrence % of Total Flt Time | Cycles in 100 Hours (n) | Oscillatory in T/E Lug (psi) | | Cycles to Failure $(N)\times10^{-6}$ | Damage Fraction (n/N) |
|---|---|---|---|---|---|---|
| $V_H$ Level Flt | | | | | | |
| 314 RPM | .0138 | | 12,551 | AB | | |
| | .2076 | | 9,556 | DA | | |
| | .1246 | | 10,647 | FC | | |
| 324 RPM | .1246 | | 10,553 | AB | | |
| | 1.8684 | | 9,234 | DA | | |
| | 1.1210 | | 10,815 | FC | | |
| Gunnery Run Pull-out | | | | | | |
| 1. To the right | | | | | | |
| c. 0.9 $V_L$ | .0100 | 191 | 17,287 | BA | 6.448 | 0.000030 |
| | .1500 | 2,871 | 18,080 | CA | 2.704 | 0.001062 |
| | .0900 | 1,723 | 17,736 | EA | 3.747 | 0.000460 |
| d. $V_L$ | .0067 | 128 | 18,851 | BA | 1.535 | 0.000083 |
| | .1002 | 1,918 | 20,080 | CA | 0.809 | 0.002371 |
| | .0601 | 1,151 | 20,152 | FA | 0.785 | 0.001466 |
| 2. To the left | | | | | | |
| c. 0.9 $V_L$ | .0100 | | 15,699 | BA | | |
| | .1500 | | 16,610 | DA | | |
| | .0900 | 1,723 | 17,705 | FA | 3.871 | 0.000445 |
| d. $V_L$ | .0067 | 128 | 18,170 | BA | 2.506 | 0.000051 |
| | .1002 | 1,918 | 18,389 | CA | 2.108 | 0.000910 |
| | .0601 | 1,151 | 19,957 | EA | 0.855 | 0.001346 |
| 3. Symmetrical | | | | | | |
| b. 0.8 $V_L$ | .0020 | | 13,832 | BA | | |
| | .0301 | | 15,656 | DA | | |
| | .0180 | 345 | 18,588 | FA | 1.826 | 0.000189 |
| c. 0.9 $V_L$ | .0033 | | 16,180 | AA | | |
| | .0501 | 959 | 17,134 | DA | 8.125 | 0.000118 |
| | .0301 | 575 | 18,733 | EA | 1.656 | 0.000347 |
| d. $V_L$ | .0007 | 13 | 19,054 | AA | 1.358 | 0.000096 |
| | .0100 | 192 | 18,698 | CA | 1.695 | 0.000113 |
| | .0060 | 115 | 18,377 | EA | 2.127 | 0.000054 |
| S - Turns | | | | | | |
| 1. 0.8 $V_H$ | .0040 | 77 | 18,385 | AA | 2.115 | 0.000036 |
| | .0600 | | 14,119 | CA | | |
| | .0360 | | 16,282 | FC | | |
| 2. $V_H$ | .0051 | 98 | 19,855 | AA | 0.896 | 0.000109 |
| | .0769 | 1,472 | 17,435 | DA | 5.290 | 0.000278 |
| | .0462 | 883 | 19,397 | FA | 1.122 | 0.000787 |
| Auto Landing | .0040 | | 6,256 | AA | | |
| | .0600 | | 11,681 | DA | | |
| | .0360 | 689 | 18,529 | FA | 1.903 | 0.000362 |

Total Damage in 100 Hr. (D) = 0.010713

Fatigue Life = 100/D = 9334 Hours

TABLE XI. 209-010-403-1 SWASHPLATE OUTER RING FATIGUE LIFE DETERMINATION

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| I.GROUND CONDITIONS | | | | | |
| A.NORMAL START | 0.4000 | 7656 | 215 AA | | 0.0 |
| B.SHUTDOWN W/COLL. | 0.4000 | 7656 | 301 AA | | 0.0 |
| II.IGE MANEUVERS | | | | | |
| A.TAKE-OFF | | | | | |
| 1.NORMAL | 0.0511 | 978 | 333 BA | | 0.0 |
| | 0.7668 | 14677 | 337 CA | | 0.0 |
| | 0.4601 | 8806 | 387 FA | | 0.0 |
| 2.JUMP | 0.0057 | 109 | 399 AA | | 0.0 |
| | 0.0852 | 1631 | 449 CA | | 0.0 |
| | 0.0511 | 978 | 574 EA | | 0.0 |
| B.HOVERING | | | | | |
| 1.STEADY | 0.0800 | 1531 | 271 BA | | 0.0 |
| | 1.2000 | 22968 | 319 DA | | 0.0 |
| | 0.7200 | 13781 | 362 FA | | 0.0 |
| 2.RIGHT TURN | 0.0067 | 128 | 284 BA | | 0.0 |
| | 0.1002 | 1918 | 362 CA | | 0.0 |
| | 0.0601 | 1151 | 362 FA | | 0.0 |
| 3.LEFT TURN | 0.0067 | 128 | 321 BA | | 0.0 |
| | 0.1002 | 1918 | 331 DA | | 0.0 |
| | 0.0601 | 1151 | 512 FA | | 0.0 |
| 4.CONTROL CORR. | | | | | |
| (A).LONGITUDINAL | 0.0007 | 13 | 506 BA | | 0.0 |
| | 0.0100 | 192 | 602 DA | | 0.0 |
| | 0.0060 | 115 | 598 EA | | 0.0 |
| (B).LATERAL | 0.0007 | 13 | 358 BA | | 0.0 |
| | 0.0100 | 192 | 637 CA | | 0.0 |
| | 0.0060 | 115 | 674 FA | | 0.0 |
| (C).RUDDER | 0.0007 | 13 | 287 AA | | 0.0 |
| | 0.0100 | 192 | 399 CA | | 0.0 |
| | 0.0060 | 115 | 323 EA | | 0.0 |
| C.SIDEWARD FLIGHT | | | | | |
| 1.TO THE RIGHT | 0.0096 | 184 | 512 AA | | 0.0 |
| | 0.1442 | 2761 | 399 CA | | 0.0 |
| | 0.0865 | 1656 | 562 FA | | 0.0 |
| 2.TO THE LEFT | 0.0096 | 184 | 395 BA | | 0.0 |
| | 0.1442 | 2761 | 442 DA | | 0.0 |
| | 0.0865 | 1656 | 449 FA | | 0.0 |
| D.REARWARD FLIGHT | 0.0096 | 184 | 543 BA | | 0.0 |
| | 0.1442 | 2761 | 550 CA | | 0.0 |
| | 0.0865 | 1656 | 682 EA | | 0.0 |
| E.ACCELERATION | | | | | |
| HOVER TO CLIMB A/S | 0.0200 | 383 | 296 BA | | 0.0 |
| | 0.3000 | 5742 | 553 DA | | 0.0 |
| | 0.1800 | 3445 | 599 FA | | 0.0 |

TABLE XI (Continued)

| FLIGHT CONDITION | | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|---|
| F.DECELERATION | | | | | | |
| 1.NORMAL | | 0.0200 | 383 | 599 AA | | 0.0 |
| | | 0.3000 | 5742 | 687 CA | | 0.0 |
| | | 0.1800 | 3445 | 634 EA | | 0.0 |
| 2.QUICK STOP | | 0.0040 | 77 | 555 BA | | 0.0 |
| | | 0.0600 | 1148 | 602 DA | | 0.0 |
| | | 0.0360 | 689 | 766 EA | | 0.0 |
| G.APPR. AND LANDING | | 0.2204 | 4218 | 605 BA | | 0.0 |
| | | 3.3057 | 63271 | 749 CA | | 0.0 |
| | | 1.9834 | 37963 | 742 EA | | 0.0 |
| III.FORWARD LEVEL FLIGHT | | | | | | |
| AIRSPEED | RPM | | | | | |
| A. 0.50 VH | 314 | 0.0104 | 196 | 400 AB | | 0.0 |
| | | 0.1563 | 2945 | 520 DA | | 0.0 |
| | | 0.0938 | 1767 | 710 FC | | 0.0 |
| | 324 | 0.0938 | 1823 | 410 AB | | 0.0 |
| | | 1.4070 | 27352 | 530 DA | | 0.0 |
| | | 0.8442 | 16411 | 520 FA | | 0.0 |
| B. 0.60 VH | 314 | 0.0309 | 582 | 485 BA | | 0.0 |
| | | 0.4634 | 8730 | 570 DA | | 0.0 |
| | | 0.2780 | 5238 | 690 FC | | 0.0 |
| | 324 | 0.2780 | 5405 | 500 AC | | 0.0 |
| | | 4.1705 | 81074 | 600 DA | | 0.0 |
| | | 2.5023 | 48644 | 620 EA | | 0.0 |
| C. 0.70 VH | 314 | 0.0342 | 644 | 570 AA | | 0.0 |
| | | 0.5131 | 9666 | 660 DA | | 0.0 |
| | | 0.3078 | 5800 | 710 FC | | 0.0 |
| | 324 | 0.3079 | 5985 | 560 AC | | 0.0 |
| | | 4.6178 | 89770 | 640 DA | | 0.0 |
| | | 2.7707 | 53862 | 760 FC | | 0.0 |
| D. 0.80 VH | 314 | 0.0551 | 1038 | 660 AB | | 0.0 |
| | | 0.8264 | 15569 | 765 DA | | 0.0 |
| | | 0.4958 | 9341 | 740 FA | | 0.0 |
| | 324 | 0.4958 | 9639 | 695 AA | | 0.0 |
| | | 7.4377 | 144588 | 705 DA | | 0.0 |
| | | 4.4626 | 86753 | 800 FC | | 0.0 |
| E. 0.90 VH | 314 | 0.0160 | 301 | 830 AB | | 0.0 |
| | | 0.2394 | 4510 | 890 DA | | 0.0 |
| | | 0.1436 | 2706 | 830 FA | | 0.0 |
| | 324 | 0.1436 | 2792 | 808 AA | | 0.0 |
| | | 2.1546 | 41885 | 805 DA | | 0.0 |
| | | 1.2928 | 25131 | 810 FC | | 0.0 |
| F. VH | 314 | 0.0138 | 261 | 1180 AB | | 0.0 |
| | | 0.2076 | 3911 | 1030 DA | | 0.0 |
| | | 0.1246 | 2347 | 950 EA | | 0.0 |

TABLE XI (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| 324 | 0.1246 | 2421 | 1090 AB | | 0.0 |
| | 1.8684 | 36322 | 920 DA | | 0.0 |
| | 1.1210 | 21793 | 930 EA | | 0.0 |
| IV.NON-FIRING MANEUVERS | | | | | |
| A.FULL POWER CLIMB | | | | | |
| 1.NORMAL | 0.1000 | 1914 | 580 AC | | 0.0 |
| | 1.5000 | 28710 | 562 CA | | 0.0 |
| | 0.9000 | 17226 | 749 FC | | 0.0 |
| 2.HIGH-SPEED | 0.0017 | 33 | 791 AC | | 0.0 |
| | 0.0256 | 489 | 1100 CA | | 0.0 |
| | 0.0153 | 294 | 848 FB | | 0.0 |
| B.MAXIMUM RATE ACCEL | | | | | |
| CLIMB - CRUISE A/S | 0.1870 | 3580 | 816 AA | | 0.0 |
| | 2.8056 | 53699 | 827 DA | | 0.0 |
| | 1.6834 | 32219 | 873 FC | | 0.0 |
| C.NORMAL TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0668 | 1279 | 481 BA | | 0.0 |
| | 1.0020 | 19178 | 599 DA | | 0.0 |
| | 0.6012 | 11507 | 749 FC | | 0.0 |
| (B) 0.7 VH | 0.0668 | 1279 | 649 AA | | 0.0 |
| | 1.0020 | 19178 | 787 DA | | 0.0 |
| | 0.6012 | 11507 | 934 FB | | 0.0 |
| (C) 0.9 VH | 0.0043 | 83 | 889 AC | | 0.0 |
| | 0.0652 | 1247 | 988 DA | | 0.0 |
| | 0.0391 | 748 | 1113 EA | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0668 | 1279 | 469 AC | | 0.0 |
| | 1.0020 | 19178 | 575 DA | | 0.0 |
| | 0.6012 | 11507 | 637 FC | | 0.0 |
| (B) 0.7 VH | 0.0668 | 1279 | 629 BA | | 0.0 |
| | 1.0020 | 19178 | 750 DA | | 0.0 |
| | 0.6012 | 11507 | 873 FB | | 0.0 |
| (C) 0.9 VH | 0.0043 | 83 | 862 AA | | 0.0 |
| | 0.0652 | 1247 | 901 DA | | 0.0 |
| | 0.0391 | 748 | 955 EA | | 0.0 |
| D. .9 VH CONTR. CORR | | | | | |
| 1.LONGITUDINAL | 0.0033 | 64 | 1299 AA | | 0.0 |
| | 0.0501 | 959 | 1037 DA | | 0.0 |
| | 0.0301 | 575 | 1094 FB | | 0.0 |
| 2.LATERAL | 0.0033 | 64 | 1511 AA | 44.086 | 0.000001 |
| | 0.0501 | 959 | 1136 DA | | 0.0 |
| | 0.0301 | 575 | 1106 FB | | 0.0 |
| 3.RUDDER | 0.0006 | 11 | 926 AB | | 0.0 |
| | 0.0085 | 162 | 926 DA | | 0.0 |

TABLE XI (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0051 | 97 | 877 FA | | 0.0 |
| E.SIDESLIP | 0.0080 | 153 | 333 AC | | 0.0 |
| | 0.1200 | 2297 | 493 DA | | 0.0 |
| | 0.0720 | 1378 | 664 FC | | 0.0 |
| F.PART POWER DESCENT | 0.0040 | 77 | 809 AB | | 0.0 |
| | 0.0600 | 1148 | 787 CA | | 0.0 |
| | 0.0360 | 689 | 869 EA | | 0.0 |
| V.GUNNERY MANEUVERS | | | | | |
| A.FIRING IN A HOVER | 0.0050 | 96 | 271 BA | | 0.0 |
| | 0.0751 | 1438 | 319 DA | | 0.0 |
| | 0.0451 | 863 | 362 FA | | 0.0 |
| B.STRAFING IN ACCEL. FROM A HOVER | 0.0033 | 64 | 296 BA | | 0.0 |
| | 0.0501 | 959 | 553 DA | | 0.0 |
| | 0.0301 | 575 | 599 FA | | 0.0 |
| C.GUNNERY RUNS | | | | | |
| 1.PT. TARGET DIVES | | | | | |
| (A) TO 0.6 VL | 0.0187 | 358 | 599 BA | | 0.0 |
| | 0.2806 | 5370 | 679 DA | | 0.0 |
| | 0.1683 | 3222 | 852 FC | | 0.0 |
| (B) TO 0.8 VL | 0.1040 | 1991 | 950 BA | | 0.0 |
| | 1.5602 | 29862 | 1025 DA | | 0.0 |
| | 0.9361 | 17917 | 919 EA | | 0.0 |
| (C) TO 0.9 VL | 0.3220 | 6164 | 1118 AA | | 0.0 |
| | 4.8305 | 92455 | 1136 DA | | 0.0 |
| | 2.8983 | 55473 | 1354 EA | | 0.0 |
| (D) TO VL | 0.0008 | 15 | 1419 AA | | 0.0 |
| | 0.0120 | 230 | 1387 CA | | 0.0 |
| | 0.0072 | 138 | 1727 EA | 9.451 | 0.000015 |
| 2.SPRAY FIRE DIVES | | | | | |
| (A) TO 0.6 VL | 0.0080 | 153 | 499 BA | | 0.0 |
| | 0.1202 | 2301 | 703 DA | | 0.0 |
| | 0.0721 | 1381 | 762 FC | | 0.0 |
| (B) TO 0.8 VL | 0.1079 | 2065 | 912 BA | | 0.0 |
| | 1.6184 | 30977 | 1000 CA | | 0.0 |
| | 0.9711 | 18586 | 1056 FA | | 0.0 |
| (C) TO 0.9 VL | 0.2291 | 4384 | 1042 AA | | 0.0 |
| | 3.4358 | 65762 | 1150 CA | | 0.0 |
| | 2.0615 | 39457 | 1354 FA | | 0.0 |
| (D) TO VL | 0.0040 | 77 | 1306 AA | | 0.0 |
| | 0.0600 | 1148 | 1337 CA | | 0.0 |
| | 0.0360 | 689 | 1615 FA | 19.216 | 0.000036 |
| D.GUNNERY RUN P/U | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.6 VL | 0.0020 | 38 | 1012 BA | | 0.0 |

## TABLE XI (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0300 | 574 | 1309 DA | | 0.0 |
| | 0.0180 | 345 | 2099 EA | 1.833 | 0.000188 |
| (B) 0.8 VL | 0.0040 | 77 | 1625 BA | 17.919 | 0.000004 |
| | 0.0600 | 1148 | 1649 CA | 15.239 | 0.000075 |
| | 0.0360 | 689 | 1888 EA | 4.202 | 0.000164 |
| (C) 0.9 VL | 0.0100 | 191 | 1557 AC | 29.756 | 0.000006 |
| | 0.1500 | 2871 | 1667 DA | 13.563 | 0.000212 |
| | 0.0900 | 1723 | 1888 EA | 4.202 | 0.000410 |
| (D) VL | 0.0067 | 128 | 1662 BA | 14.004 | 0.000009 |
| | 0.1002 | 1918 | 1825 CA | 5.638 | 0.000340 |
| | 0.0601 | 1151 | 1963 EA | 3.053 | 0.000377 |
| 2.TO THE LEFT | | | | | |
| (A) 0.6 VL | 0.0020 | 38 | 1200 BA | | 0.0 |
| | 0.0300 | 574 | 1299 CA | | 0.0 |
| | 0.0180 | 345 | 1913 EA | 3.764 | 0.000092 |
| (B) 0.8 VL | 0.0040 | 77 | 1537 BA | 35.102 | 0.000002 |
| | 0.0600 | 1148 | 1864 DA | 4.685 | 0.000245 |
| | 0.0360 | 689 | 2101 FB | 1.820 | 0.000379 |
| (C) 0.9 VL | 0.0100 | 191 | 2012 BA | 2.517 | 0.000076 |
| | 0.1500 | 2871 | 1887 CA | 4.220 | 0.000680 |
| | 0.0900 | 1723 | 1888 EA | 4.202 | 0.000410 |
| (D) VL | 0.0067 | 128 | 1532 AA | 36.631 | 0.000003 |
| | 0.1002 | 1918 | 1729 DA | 9.344 | 0.000205 |
| | 0.0601 | 1151 | 1940 FB | 3.357 | 0.000343 |
| 3.SYMMETRICAL | | | | | |
| (A) 0.6 VL | 0.0002 | 4 | 937 BA | | 0.0 |
| | 0.0030 | 57 | 1148 DA | | 0.0 |
| | 0.0018 | 34 | 1416 FA | | 0.0 |
| (B) 0.8 VL | 0.0020 | 38 | 1224 BA | | 0.0 |
| | 0.0301 | 575 | 1556 DA | 29.997 | 0.000019 |
| | 0.0180 | 345 | 1876 FA | 4.435 | 0.000078 |
| (C) 0.9 VL | 0.0033 | 64 | 1545 AC | 32.825 | 0.000002 |
| | 0.0501 | 959 | 1654 DA | 14.748 | 0.000065 |
| | 0.0301 | 575 | 2076 FB | 1.987 | 0.000290 |
| (D) VL | 0.0007 | 13 | 1631 AC | 17.195 | 0.000001 |
| | 0.0100 | 192 | 1662 CA | 14.004 | 0.000014 |
| | 0.0060 | 115 | 1820 FC | 5.778 | 0.000020 |
| E.GUNNERY TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0250 | 479 | 840 AC | | 0.0 |
| | 0.3757 | 7191 | 724 DA | | 0.0 |
| | 0.2254 | 4315 | 799 FC | | 0.0 |
| (B) 0.7 VH | 0.0729 | 1396 | 1223 AC | | 0.0 |
| | 1.09[illegible]2 | 20942 | 1296 DA | | 0.0 |
| | 0.6565 | 12565 | 1421 FA | | 0.0 |
| (C) 0.9 VH | 0.0016 | 31 | 1483 AC | 57.421 | 0.000001 |

TABLE XI (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0240 | 459 | 1457 DA | 74.904 | 0.000006 |
| | 0.0144 | 276 | 1790 EA | 6.719 | 0.000041 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0250 | 479 | 703 BA | | 0.0 |
| | 0.3757 | 7191 | 712 CA | | 0.0 |
| | 0.2254 | 4315 | 737 FB | | 0.0 |
| (B) 0.7 VH | 0.0729 | 1396 | 902 AC | | 0.0 |
| | 1.0942 | 20942 | 1099 DA | | 0.0 |
| | 0.6565 | 12565 | 1137 EA | | 0.0 |
| (C) 0.9 VH | 0.0016 | 31 | 1792 AC | 6.651 | 0.000005 |
| | 0.0240 | 459 | 1432 DA | 98.848 | 0.000005 |
| | 0.0144 | 276 | 1911 EA | 3.797 | 0.000073 |
| F. S-TURNS | | | | | |
| 1.AT 0.8 VH | 0.0040 | 77 | 1449 AA | 81.652 | 0.000001 |
| | 0.0600 | 1148 | 1321 DA | | 0.0 |
| | 0.0360 | 689 | 1685 FC | 12.120 | 0.000057 |
| 2.AT VH | 0.0051 | 98 | 1774 AA | 7.304 | 0.000013 |
| | 0.0769 | 1472 | 1519 DA | 41.031 | 0.000036 |
| | 0.0462 | 883 | 1746 FC | 8.495 | 0.000104 |
| VI.POWER TRANSITIONS | | | | | |
| A.POWER TO AUTO | | | | | |
| 1. 0.5 VH | 0.0033 | 64 | 456 BA | | 0.0 |
| | 0.0501 | 959 | 543 DA | | 0.0 |
| | 0.0301 | 575 | 608 FA | | 0.0 |
| 2. 0.7 VH | 0.0083 | 160 | 629 AB | | 0.0 |
| | 0.1252 | 2397 | 712 CA | | 0.0 |
| | 0.0751 | 1438 | 762 FB | | 0.0 |
| 3. 0.9 VH | 0.0060 | 115 | 839 AB | | 0.0 |
| | 0.0900 | 1723 | 852 DA | | 0.0 |
| | 0.0540 | 1034 | 889 FB | | 0.0 |
| B.AUTO TO POWER | | | | | |
| 1.IN GROUND-EFFECT | 0.0040 | 77 | 587 AA | | 0.0 |
| | 0.0600 | 1148 | 651 DA | | 0.0 |
| | 0.0360 | 689 | 799 FA | | 0.0 |
| 2. 0.4 VH | 0.0067 | 128 | 592 BA | | 0.0 |
| | 0.1002 | 1918 | 716 DA | | 0.0 |
| | 0.0601 | 1151 | 639 FB | | 0.0 |
| 3. 0.6 VH | 0.0324 | 620 | 926 AB | | 0.0 |
| | 0.4862 | 9307 | 937 CA | | 0.0 |
| | 0.2917 | 5584 | 1180 FB | | 0.0 |
| 4. MAX AUTO A/S | 0.0012 | 23 | 1086 BA | | 0.0 |
| | 0.0180 | 345 | 1136 DA | | 0.0 |
| | 0.0108 | 207 | 1643 FB | 15.857 | 0.000013 |
| VII.AUTOROTATION | | | | | |

TABLE XI (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY PITCH LINK AXIAL LOAD | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| A. STABILIZED FLIGHT | | | | | |
| 1. 0.4 VH | 0.0043 | 83 | 271 BA | | 0.0 |
| | 0.0652 | 1247 | 444 DA | | 0.0 |
| | 0.0391 | 748 | 298 FA | | 0.0 |
| 2. 0.6 VH | 0.0809 | 1549 | 395 BA | | 0.0 |
| | 1.2142 | 23239 | 580 DA | | 0.0 |
| | 0.7285 | 13943 | 549 FC | | 0.0 |
| 3. MAX AUTO A/S | 0.0040 | 77 | 662 AA | | 0.0 |
| | 0.0600 | 1148 | 703 DA | | 0.0 |
| | 0.0360 | 689 | 889 FC | | 0.0 |
| B. AUTO TURNS | | | | | |
| 1. TO THE RIGHT | | | | | |
| (A) 0.4 VH | 0.0033 | 64 | 494 BA | | 0.0 |
| | 0.0501 | 959 | 568 DA | | 0.0 |
| | 0.0301 | 575 | 521 EA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 1036 | 629 BA | | 0.0 |
| | 0.8119 | 15540 | 716 DA | | 0.0 |
| | 0.4872 | 9324 | 716 FC | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 38 | 975 AB | | 0.0 |
| | 0.0300 | 574 | 1111 DA | | 0.0 |
| | 0.0180 | 345 | 1590 EA | 23.040 | 0.000015 |
| 2. TO THE LEFT | | | | | |
| (A) 0.4 VH | 0.0033 | 64 | 494 BA | | 0.0 |
| | 0.0501 | 959 | 592 DA | | 0.0 |
| | 0.0301 | 575 | 491 FB | | 0.0 |
| (B) 0.6 VH | 0.0541 | 1036 | 618 AC | | 0.0 |
| | 0.8119 | 15540 | 765 DA | | 0.0 |
| | 0.4872 | 9324 | 646 FA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 38 | 987 AB | | 0.0 |
| | 0.0300 | 574 | 1111 DA | | 0.0 |
| | 0.0180 | 345 | 1577 EA | 25.423 | 0.000014 |
| C. AUTO LANDING | 0.0040 | 77 | 802 BA | | 0.0 |
| | 0.0600 | 1148 | 799 DA | | 0.0 |
| | 0.0360 | 689 | 1049 FA | | 0.0 |

ENDURANCE LIMIT = 1431.0 TOTAL DAMAGE (D) = 0.005143
MATERIAL = ALUM
FREQUENCY = 1 / REV OF M/R FATIGUE LIFE = 100/D = 19443 HOURS

TABLE XII. 204-011-702-17 TAIL ROTOR BLADE FATIGUE LIFE DETERMINATION

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| I.GROUND CONDITIONS | | | | | |
| A.NORMAL START | 0.4000 | 15312 | 1107 AA | | 0.0 |
| B.SHUTDOWN W/COLL. | 0.4000 | 15312 | 1825 AA | | 0.0 |
| II.IGE MANEUVERS | | | | | |
| A.TAKE-OFF | | | | | |
| 1.NORMAL | 0.0511 | 1957 | 1631 AA | | 0.0 |
| | 0.7668 | 29353 | 2038 CA | | 0.0 |
| | 0.4601 | 17612 | 1988 FA | | 0.0 |
| 2.JUMP | 0.0057 | 217 | 2141 BA | | 0.0 |
| | 0.0852 | 3261 | 2233 CA | | 0.0 |
| | 0.0511 | 1957 | 1148 FA | | 0.0 |
| B.HOVERING | | | | | |
| 1.STEADY | 0.0800 | 3062 | 1684 AA | | 0.0 |
| | 1.2000 | 45936 | 1909 CA | | 0.0 |
| | 0.7200 | 27562 | 2240 FA | | 0.0 |
| 2.RIGHT TURN | 0.0067 | 256 | 2259 BA | | 0.0 |
| | 0.1002 | 3836 | 2756 CA | | 0.0 |
| | 0.0601 | 2301 | 2734 EA | | 0.0 |
| 3.LEFT TURN | 0.0067 | 256 | 3203 BA | | 0.0 |
| | 0.1002 | 3836 | 2485 CA | | 0.0 |
| | 0.0601 | 2301 | 2305 EA | | 0.0 |
| 4.CONTROL CORR. | | | | | |
| (A).LONGITUDINAL | 0.0007 | 26 | 2235 BA | | 0.0 |
| | 0.0100 | 384 | 2027 CA | | 0.0 |
| | 0.0060 | 230 | 1964 FA | | 0.0 |
| (B).LATERAL | 0.0007 | 26 | 1979 AA | | 0.0 |
| | 0.0100 | 384 | 2537 CA | | 0.0 |
| | 0.0060 | 230 | 2246 FA | | 0.0 |
| (C).RUDDER | 0.0007 | 26 | 2431 BA | | 0.0 |
| | 0.0100 | 384 | 2139 CA | | 0.0 |
| | 0.0060 | 230 | 2275 EA | | 0.0 |
| C.SIDEWARD FLIGHT | | | | | |
| 1.TO THE RIGHT | 0.0096 | 368 | 3217 BA | | 0.0 |
| | 0.1442 | 5522 | 2822 CA | | 0.0 |
| | 0.0865 | 3313 | 2888 FA | | 0.0 |
| 2.TO THE LEFT | 0.0096 | 368 | 748 AA | | 0.0 |
| | 0.1442 | 5522 | 537 CA | | 0.0 |
| | 0.0865 | 3313 | 520 FA | | 0.0 |
| D.REARWARD FLIGHT | 0.0096 | 368 | 1508 AA | | 0.0 |
| | 0.1442 | 5522 | 1598 CA | | 0.0 |
| | 0.0865 | 3313 | 2383 FA | | 0.0 |
| E.ACCELERATION | | | | | |
| HOVER TO CLIMB A/S | 0.0200 | 766 | 1864 BA | | 0.0 |
| | 0.3000 | 11484 | 2862 CA | | 0.0 |
| | 0.1800 | 6890 | 2305 FA | | 0.0 |

TABLE XII (Continued)

T/R BLADE

| FLIGHT CONDITION | | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|---|
| F.DECELERATION | | | | | | |
| 1.NORMAL | | 0.0200 | 766 | 1965 BA | | 0.0 |
| | | 0.3000 | 11484 | 2686 CA | | 0.0 |
| | | 0.1800 | 6890 | 2498 FA | | 0.0 |
| 2.QUICK STOP | | 0.0040 | 153 | 1901 BA | | 0.0 |
| | | 0.0600 | 2297 | 2315 CA | | 0.0 |
| | | 0.0360 | 1378 | 3010 FA | | 0.0 |
| G.APPR. AND LANDING | | 0.2204 | 8436 | 2054 AA | | 0.0 |
| | | 3.3057 | 126542 | 2099 CA | | 0.0 |
| | | 1.9834 | 75925 | 2054 EA | | 0.0 |
| III.FORWARD LEVEL FLIGHT | | | | | | |
| AIRSPEED | RPM | | | | | |
| A. 0.50 VH | 314 | 0.0104 | 393 | 1010 AA | | 0.0 |
| | | 0.1563 | 5889 | 1300 DA | | 0.0 |
| | | 0.0938 | 3534 | 1187 FA | | 0.0 |
| | 324 | 0.0938 | 3647 | 910 AA | | 0.0 |
| | | 1.4070 | 54704 | 981 DA | | 0.0 |
| | | 0.8442 | 32822 | 1162 FB | | 0.0 |
| B. 0.60 VH | 314 | 0.0309 | 1164 | 1152 BA | | 0.0 |
| | | 0.4634 | 17460 | 1246 DA | | 0.0 |
| | | 0.2760 | 10476 | 1528 FA | | 0.0 |
| | 324 | 0.2780 | 10810 | 1295 AA | | 0.0 |
| | | 4.1705 | 162.48 | 1126 DA | | 0.0 |
| | | 2.5023 | 97289 | 1468 FB | | 0.0 |
| C. 0.70 VH | 314 | 0.0342 | 1289 | 1599 AA | | 0.0 |
| | | 0.5131 | 19332 | 1373 DA | | 0.0 |
| | | 0.3078 | 11599 | 1956 EA | | 0.0 |
| | 324 | 0.3079 | 11969 | 1606 AA | | 0.0 |
| | | 4.6178 | 179539 | 1449 CA | | 0.0 |
| | | 2.7707 | 107724 | 1803 FB | | 0.0 |
| D. 0.80 VH | 314 | 0.0551 | 2076 | 2168 BA | | 0.0 |
| | | 0.8264 | 31138 | 1704 DA | | 0.0 |
| | | 0.4958 | 18683 | 2250 EA | | 0.0 |
| | 324 | 0.4958 | 19278 | 2178 AA | | 0.0 |
| | | 7.4377 | 289176 | 1927 DA | | 0.0 |
| | | 4.4626 | 173506 | 2262 FB | | 0.0 |
| E. 0.90 VH | 314 | 0.0160 | 601 | 2220 AA | | 0.0 |
| | | 0.2394 | 9021 | 2281 DA | | 0.0 |
| | | 0.1436 | 5412 | 2967 EA | | 0.0 |
| | 324 | 0.1436 | 5585 | 2394 BA | | 0.0 |
| | | 2.1546 | 83771 | 2583 DA | | 0.0 |
| | | 1.2928 | 50262 | 2873 EA | | 0.0 |
| F. VH | 314 | 0.0138 | 521 | 2632 AA | | 0.0 |
| | | 0.2076 | 7822 | 3414 CA | 70.530 | 0.000111 |
| | | 0.1246 | 4693 | 3395 FB | 76.879 | 0.000061 |

TABLE XII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| 324 | 0.1246 | 4843 | 3288 AC | | 0.0 |
| | 1.8684 | 72643 | 2952 CA | | 0.0 |
| | 1.1210 | 43586 | 3485 FB | 52.060 | 0.000837 |
| IV.NON-FIRING MANEUVERS | | | | | |
| A.FULL POWER CLIMB | | | | | |
| 1.NORMAL | 0.1000 | 3828 | 1755 AC | | 0.0 |
| | 1.5000 | 57420 | 3181 DA | | 0.0 |
| | 0.9000 | 34452 | 2084 EA | | 0.0 |
| 2.HIGH-SPEED | 0.0017 | 65 | 2968 AC | | 0.0 |
| | 0.0256 | 978 | 3199 DA | | 0.0 |
| | 0.0153 | 587 | 3281 EA | | 0.0 |
| B.MAXIMUM RATE ACCEL | | | | | |
| CLIMB - CRUISE A/S | 0.1870 | 7160 | 2879 AC | | 0.0 |
| | 2.8056 | 107398 | 3231 DA | | 0.0 |
| | 1.6834 | 64439 | 2966 EA | | 0.0 |
| C.NORMAL TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0668 | 2557 | 1247 AA | | 0.0 |
| | 1.0020 | 38356 | 1269 FB | | 0.0 |
| | 0.6012 | 23014 | 1331 FB | | 0.0 |
| (B) 0.7 VH | 0.0668 | 2557 | 1992 AA | | 0.0 |
| | 1.0020 | 38356 | 1831 CA | | 0.0 |
| | 0.6012 | 23014 | 2028 FB | | 0.0 |
| (C) 0.9 VH | 0.0043 | 166 | 2611 AC | | 0.0 |
| | 0.0652 | 2494 | 3281 EA | | 0.0 |
| | 0.0391 | 1497 | 3133 FB | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0668 | 2557 | 1286 AC | | 0.0 |
| | 1.0020 | 38356 | 1347 DA | | 0.0 |
| | 0.6012 | 23014 | 1451 FC | | 0.0 |
| (B) 0.7 VH | 0.0668 | 2557 | 2093 AB | | 0.0 |
| | 1.0020 | 38356 | 2140 DA | | 0.0 |
| | 0.6012 | 23014 | 2436 EA | | 0.0 |
| (C) 0.9 VH | 0.0043 | 166 | 3096 AB | | 0.0 |
| | 0.0652 | 2494 | 3860 DA | 14.768 | 0.000169 |
| | 0.0391 | 1497 | 3387 EA | 79.775 | 0.000019 |
| D. .9 VH CONTR. CORR | | | | | |
| 1.LONGITUDINAL | 0.0033 | 128 | 2595 BA | | 0.0 |
| | 0.0501 | 1918 | 3191 DA | | 0.0 |
| | 0.0301 | 1151 | 2978 FB | | 0.0 |
| 2.LATERAL | 0.0033 | 128 | 3269 AA | | 0.0 |
| | 0.0501 | 1918 | 3334 DA | | 0.0 |
| | 0.0301 | 1151 | 3381 EA | 82.040 | 0.000014 |
| 3.RUDDER | 0.0006 | 22 | 4584 AA | 3.041 | 0.000007 |
| | 0.0085 | 324 | 3227 DA | | 0.0 |

TABLE XII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0051 | 194 | 3146 FB | | 0.0 |
| E.SIDESLIP | 0.0080 | 306 | 904 AC | | 0.0 |
| | 0.1200 | 4594 | 1165 DA | | 0.0 |
| | 0.0720 | 2756 | 1600 FC | | 0.0 |
| F.PART POWER DESCENT | 0.0040 | 153 | 2383 BA | | 0.0 |
| | 0.0600 | 2297 | 2421 CA | | 0.0 |
| | 0.0360 | 1378 | 2141 FA | | 0.0 |
| V.GUNNERY MANEUVERS | | | | | |
| A.FIRING IN A HOVER | 0.0050 | 192 | 1684 AA | | 0.0 |
| | 0.0751 | 2876 | 1909 CA | | 0.0 |
| | 0.0451 | 1725 | 2240 FA | | 0.0 |
| B.STRAFING IN ACCEL. FROM A HOVER | 0.0033 | 128 | 1864 BA | | 0.0 |
| | 0.0501 | 1918 | 2862 CA | | 0.0 |
| | 0.0301 | 1151 | 2305 FA | | 0.0 |
| C.GUNNERY RUNS | | | | | |
| 1.PT. TARGET DIVES | | | | | |
| (A) TO 0.6 VL | 0.0187 | 716 | 1554 AB | | 0.0 |
| | 0.2806 | 10740 | 1729 CA | | 0.0 |
| | 0.1683 | 6444 | 2135 FA | | 0.0 |
| (B) TO 0.8 VL | 0.1040 | 3982 | 2154 BA | | 0.0 |
| | 1.5602 | 59724 | 2632 DA | | 0.0 |
| | 0.9361 | 35834 | 2588 EA | | 0.0 |
| (C) TO 0.9 VL | 0.3220 | 12327 | 3002 AA | | 0.0 |
| | 4.8305 | 184911 | 2960 CA | | 0.0 |
| | 2.8983 | 110946 | 3038 FA | | 0.0 |
| (D) TO VL | 0.0008 | 31 | 3291 AA | | 0.0 |
| | 0.0120 | 459 | 3719 DA | 22.468 | 0.000020 |
| | 0.0072 | 276 | 4057 EA | 8.872 | 0.000031 |
| 2.SPRAY FIRE DIVES | | | | | |
| (A) TO 0.6 VL | 0.0080 | 307 | 1666 BA | | 0.0 |
| | 0.1202 | 4603 | 2203 DA | | 0.0 |
| | 0.0721 | 2762 | 2217 FA | | 0.0 |
| (B) TO 0.8 VL | 0.1079 | 4130 | 2200 BA | | 0.0 |
| | 1.6184 | 61954 | 2837 CA | | 0.0 |
| | 0.9711 | 37172 | 2500 EA | | 0.0 |
| (C) TO 0.9 VL | 0.2291 | 8768 | 2782 BA | | 0.0 |
| | 3.4358 | 131524 | 3424 CA | 67.461 | 0.001950 |
| | 2.0615 | 78914 | 3104 FA | | 0.0 |
| (D) TO VL | 0.0040 | 153 | 3228 AB | | 0.0 |
| | 0.0600 | 2297 | 3381 CA | 82.040 | 0.000028 |
| | 0.0360 | 1378 | 4293 FC | 5.262 | 0.000262 |
| D.GUNNERY RUN P/U | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.6 VL | 0.0020 | 77 | 1718 AB | | 0.0 |

TABLE XII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0300 | 1148 | 1940 DA | | 0.0 |
| | 0.0180 | 689 | 2616 FA | | 0.0 |
| (B) 0.8 VL | 0.0040 | 153 | 3089 AA | | 0.0 |
| | 0.0600 | 2297 | 3399 CA | 75.483 | 0.000030 |
| | 0.0360 | 1378 | 4193 EA | 6.501 | 0.000212 |
| (C) 0.9 VL | 0.0100 | 383 | 4261 BA | 5.622 | 0.000068 |
| | 0.1500 | 5742 | 3611 CA | 32.257 | 0.000178 |
| | 0.0900 | 3445 | 4629 EA | 2.815 | 0.001224 |
| (D) VL | 0.0067 | 256 | 3586 BA | 35.282 | 0.000007 |
| | 0.1002 | 3836 | 4641 CA | 2.759 | 0.001390 |
| | 0.0601 | 2301 | 4959 EA | 1.678 | 0.001371 |
| 2.TO THE LEFT | | | | | |
| (A) 0.6 VL | 0.0020 | 77 | 2142 BA | | 0.0 |
| | 0.0300 | 1148 | 2779 CA | | 0.0 |
| | 0.0180 | 689 | 2466 FB | | 0.0 |
| (B) 0.8 VL | 0.0040 | 153 | 3563 BA | 38.399 | 0.000004 |
| | 0.0600 | 2297 | 3218 FA | | 0.0 |
| | 0.0360 | 1378 | 3836 FA | 15.804 | 0.000087 |
| (C) 0.9 VL | 0.0100 | 383 | 5182 BA | 1.232 | 0.000311 |
| | 0.1500 | 5742 | 4420 CA | 4.096 | 0.001402 |
| | 0.0900 | 3445 | 4475 EA | 3.696 | 0.000932 |
| (D) VL | 0.0067 | 256 | 4812 BA | 2.092 | 0.000122 |
| | 0.1002 | 3836 | 4842 CA | 1.998 | 0.001920 |
| | 0.0601 | 2301 | 5325 EA | 1.025 | 0.002245 |
| 3.SYMMETRICAL | | | | | |
| (A) 0.6 VL | 0.0002 | 8 | 1728 AA | | 0.0 |
| | 0.0030 | 115 | 2042 DA | | 0.0 |
| | 0.0018 | 69 | 2462 FA | | 0.0 |
| (B) 0.8 VL | 0.0020 | 77 | 3603 BA | 33.187 | 0.000002 |
| | 0.0301 | 1151 | 3639 CA | 29.257 | 0.000039 |
| | 0.0180 | 690 | 3975 EA | 10.866 | 0.000064 |
| (C) 0.9 VL | 0.0033 | 128 | 3821 BA | 16.499 | 0.000008 |
| | 0.0501 | 1918 | 3838 CA | 15.714 | 0.000122 |
| | 0.0301 | 1151 | 4759 EA | 2.274 | 0.000506 |
| (D) VL | 0.0007 | 26 | 4461 BA | 3.792 | 0.000007 |
| | 0.0100 | 384 | 4273 CA | 5.483 | 0.000070 |
| | 0.0060 | 230 | 4604 FB | 2.938 | 0.000078 |
| E.GUNNERY TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0250 | 959 | 1432 AB | | 0.0 |
| | 0.3757 | 14383 | 1436 CA | | 0.0 |
| | 0.2254 | 8630 | 1641 FB | | 0.0 |
| (B) 0.7 VH | 0.0729 | 2792 | 2307 AB | | 0.0 |
| | 1.0942 | 41884 | 2085 DA | | 0.0 |
| | 0.6565 | 25131 | 2418 FC | | 0.0 |
| (C) 0.9 VH | 0.0016 | 61 | 2957 AC | | 0.0 |

TABLE XII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0240 | 919 | 3085 DA | | 0.0 |
| | 0.0144 | 551 | 2967 FB | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0250 | 959 | 2010 AB | | 0.0 |
| | 0.3757 | 14383 | 1565 CA | | 0.0 |
| | 0.2254 | 8630 | 1900 FA | | 0.0 |
| (B) 0.7 VH | 0.0729 | 2792 | 2373 BA | | 0.0 |
| | 1.0942 | 41884 | 2387 DA | | 0.0 |
| | 0.6565 | 25131 | 2778 FA | | 0.0 |
| (C) 0.9 VH | 0.0016 | 61 | 3953 AB | 11.498 | 0.000005 |
| | 0.0240 | 919 | 3340 DA | 100.000 | 0.000009 |
| | 0.0144 | 551 | 4226 EA | 6.053 | 0.000091 |
| F. S-TURNS | | | | | |
| 1.AT 0.8 VH | 0.0040 | 153 | 3428 AA | 66.282 | 0.000002 |
| | 0.0600 | 2297 | 3880 CA | 13.971 | 0.000164 |
| | 0.0360 | 1378 | 3518 FA | 45.614 | 0.000030 |
| 2.AT VH | 0.0051 | 196 | 4754 BA | 2.292 | 0.000086 |
| | 0.0769 | 2944 | 4825 DA | 2.051 | 0.001436 |
| | 0.0462 | 1767 | 5789 FB | 0.600 | 0.002943 |
| VI.POWER TRANSITIONS | | | | | |
| A.POWER TO AUTO | | | | | |
| 1. 0.5 VH | 0.0033 | 128 | 1601 AA | | 0.0 |
| | 0.0501 | 1918 | 1184 DA | | 0.0 |
| | 0.0301 | 1151 | 1331 FA | | 0.0 |
| 2. 0.7 VH | 0.0083 | 320 | 2171 AA | | 0.0 |
| | 0.1252 | 4793 | 1646 DA | | 0.0 |
| | 0.0751 | 2876 | 2129 EA | | 0.0 |
| 3. 0.9 VH | 0.0060 | 230 | 2861 BA | | 0.0 |
| | 0.0900 | 3445 | 2155 DA | | 0.0 |
| | 0.0540 | 2067 | 3260 FB | | 0.0 |
| B.AUTO TO POWER | | | | | |
| 1.IN GROUND-EFFECT | 0.0040 | 153 | 1829 AA | | 0.0 |
| | 0.0600 | 2297 | 2175 CA | | 0.0 |
| | 0.0360 | 1378 | 2090 EA | | 0.0 |
| 2. 0.4 VH | 0.0067 | 256 | 1630 AB | | 0.0 |
| | 0.1002 | 3836 | 1848 DA | | 0.0 |
| | 0.0601 | 2301 | 1892 FB | | 0.0 |
| 3. 0.6 VH | 0.0324 | 1241 | 2131 AB | | 0.0 |
| | 0.4862 | 18613 | 2339 DA | | 0.0 |
| | 0.2917 | 11168 | 2474 FB | | 0.0 |
| 4. MAX AUTO A/S | 0.0012 | 46 | 3332 AB | | 0.0 |
| | 0.0180 | 689 | 2641 DA | | 0.0 |
| | 0.0108 | 413 | 3440 FB | 62.903 | 0.000007 |
| VII.AUTOROTATION | | | | | |

TABLE XII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY T/R BLADE SKIN STRESS STA. 21.5 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| A.STABILIZED FLIGHT | | | | | |
| 1. 0.4 VH | 0.0043 | 166 | 1446 AA | | 0.0 |
| | 0.0652 | 2494 | 771 DA | | 0.0 |
| | 0.0391 | 1497 | 994 FA | | 0.0 |
| 2. 0.6 VH | 0.0809 | 3099 | 1400 AA | | 0.0 |
| | 1.2142 | 46478 | 1084 DA | | 0.0 |
| | 0.7285 | 27887 | 1135 FA | | 0.0 |
| 3. MAX AUTO A/S | 0.0040 | 153 | 2836 AA | | 0.0 |
| | 0.0600 | 2297 | 1914 DA | | 0.0 |
| | 0.0360 | 1378 | 2518 FA | | 0.0 |
| B.AUTO TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.4 VH | 0.0033 | 128 | 2025 AA | | 0.0 |
| | 0.0501 | 1918 | 1143 DA | | 0.0 |
| | 0.0301 | 1151 | 1074 EA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 2072 | 1772 AB | | 0.0 |
| | 0.8119 | 31080 | 1131 DA | | 0.0 |
| | 0.4872 | 18648 | 1385 FA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 77 | 2653 AC | | 0.0 |
| | 0.0300 | 1148 | 2661 CA | | 0.0 |
| | 0.0180 | 689 | 3396 EA | 76.527 | 0.000009 |
| 2.TO THE LEFT | | | | | |
| (A) 0.4 VH | 0.0033 | 128 | 1228 BA | | 0.0 |
| | 0.0501 | 1918 | 1067 CA | | 0.0 |
| | 0.0301 | 1151 | 1328 EA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 2072 | 1481 AC | | 0.0 |
| | 0.8119 | 31080 | 1436 CA | | 0.0 |
| | 0.4872 | 18648 | 1415 FA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 77 | 2992 AA | | 0.0 |
| | 0.0300 | 1148 | 2483 CA | | 0.0 |
| | 0.0180 | 689 | 2789 EA | | 0.0 |
| C.AUTO LANDING | 0.0040 | 153 | 1529 BA | | 0.0 |
| | 0.0600 | 2297 | 3194 CA | | 0.0 |
| | 0.0360 | 1378 | 3428 EA | 66.282 | 0.000021 |

ENDURANCE LIMIT = 3340.0 TOTAL DAMAGE (D) = 0.020713
MATERIAL = ALUM
FREQUENCY = 2 / REV OF M/R FATIGUE LIFE = 100/D = 4827 HOURS

TABLE XIII. 204-011-728-5 TAIL ROTOR GRIP FATIGUE LIFE DETERMINATION

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| I.GROUND CONDITIONS | | | | | |
| A.NORMAL START | 0.4000 | 15373 | 1090 CA | | 0.0 |
| B.SHUTDOWN W/COLL. | 0.4000 | 15373 | 1291 CA | | 0.0 |
| II.IGE MANEUVERS | | | | | |
| A.TAKE-OFF | | | | | |
| 1.NORMAL | 0.0511 | 1965 | 813 AA | | 0.0 |
| | 0.7668 | 29471 | 1024 CA | | 0.0 |
| | 0.4601 | 17682 | 979 FA | | 0.0 |
| 2.JUMP | 0.0057 | 218 | 841 AA | | 0.0 |
| | 0.0852 | 3274 | 1520 CA | | 0.0 |
| | 0.0511 | 1965 | 1884 FA | | 0.0 |
| B.HOVERING | | | | | |
| 1.STEADY | 0.0800 | 3075 | 631 AA | | 0.0 |
| | 1.2000 | 46120 | 799 CA | | 0.0 |
| | 0.7200 | 27672 | 887 FA | | 0.0 |
| 2.RIGHT TURN | 0.0067 | 257 | 1739 AA | | 0.0 |
| | 0.1002 | 3851 | 1612 CA | | 0.0 |
| | 0.0601 | 2311 | 1249 FA | | 0.0 |
| 3.LEFT TURN | 0.0067 | 257 | 772 AA | | 0.0 |
| | 0.1002 | 3851 | 1137 DA | | 0.0 |
| | 0.0601 | 2311 | 882 FA | | 0.0 |
| 4.CONTROL CORR. | | | | | |
| (A).LONGITUDINAL | 0.0007 | 26 | 946 AA | | 0.0 |
| | 0.0100 | 385 | 979 CA | | 0.0 |
| | 0.0060 | 231 | 1038 FA | | 0.0 |
| (B).LATERAL | 0.0007 | 26 | 869 AA | | 0.0 |
| | 0.0100 | 385 | 1074 DA | | 0.0 |
| | 0.0060 | 231 | 940 FA | | 0.0 |
| (C).RUDDER | 0.0007 | 26 | 948 AA | | 0.0 |
| | 0.0100 | 385 | 1160 CA | | 0.0 |
| | 0.0060 | 231 | 921 FA | | 0.0 |
| C.SIDEWARD FLIGHT | | | | | |
| 1.TO THE RIGHT | 0.0096 | 370 | 1589 AA | | 0.0 |
| | 0.1442 | 5544 | 1708 DA | | 0.0 |
| | 0.0865 | 3326 | 1371 FA | | 0.0 |
| 2.TO THE LEFT | 0.0096 | 370 | 1255 AA | | 0.0 |
| | 0.1442 | 5544 | 1881 CA | | 0.0 |
| | 0.0865 | 3326 | 2556 FA | | 0.0 |
| D.REARWARD FLIGHT | 0.0096 | 370 | 1411 AA | | 0.0 |
| | 0.1442 | 5544 | 5082 CA | 0.664 | 0.008350 |
| | 0.0865 | 3326 | 1920 FA | | 0.0 |
| E.ACCELERATION | | | | | |
| HOVER TO CLIMB A/S | 0.0200 | 769 | 1270 AA | | 0.0 |
| | 0.3000 | 11530 | 1193 CA | | 0.0 |
| | 0.1800 | 6918 | 1603 FA | | 0.0 |

TABLE XIII (Continued)

| FLIGHT CONDITION | | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|---|
| F.DECELERATION | | | | | | |
| 1.NORMAL | | 0.0200 | 769 | 1466 AA | | 0.0 |
| | | 0.3000 | 11530 | 1256 DA | | 0.0 |
| | | 0.1800 | 6918 | 1413 FA | | 0.0 |
| 2.QUICK STOP | | 0.0040 | 154 | 1460 AA | | 0.0 |
| | | 0.0600 | 2306 | 1298 CA | | 0.0 |
| | | 0.0360 | 1384 | 1958 FA | | 0.0 |
| G.APPR. AND LANDING | | 0.2204 | 8470 | 1226 AA | | 0.0 |
| | | 3.3057 | 127048 | 1315 CA | | 0.0 |
| | | 1.9834 | 76229 | 1892 FA | | 0.0 |
| III.FORWARD LEVEL FLIGHT | | | | | | |
| AIRSPEED | RPM | | | | | |
| A. 0.50 VH | 314 | 0.0104 | 394 | 946 AA | | 0.0 |
| | | 0.1563 | 5913 | 962 BB | | 0.0 |
| | | 0.0938 | 3548 | 1244 FA | | 0.0 |
| | 324 | 0.0938 | 3662 | 1027 AA | | 0.0 |
| | | 1.4070 | 54923 | 1031 BB | | 0.0 |
| | | 0.8442 | 32954 | 1182 FA | | 0.0 |
| B. 0.60 VH | 314 | 0.0309 | 1169 | 1242 AA | | 0.0 |
| | | 0.4634 | 17530 | 1218 BB | | 0.0 |
| | | 0.2780 | 10518 | 1284 FA | | 0.0 |
| | 324 | 0.2780 | 10853 | 1107 AA | | 0.0 |
| | | 4.1705 | 162797 | 1195 BB | | 0.0 |
| | | 2.5023 | 97678 | 1294 FA | | 0.0 |
| C. 0.70 VH | 314 | 0.0342 | 1294 | 1309 AA | | 0.0 |
| | | 0.5131 | 19409 | 1501 BB | | 0.0 |
| | | 0.3078 | 11646 | 1410 FA | | 0.0 |
| | 324 | 0.3079 | 12017 | 1400 AA | | 0.0 |
| | | 4.6178 | 180257 | 1379 BB | | 0.0 |
| | | 2.7707 | 108154 | 1463 FA | | 0.0 |
| D. 0.80 VH | 314 | 0.0551 | 2084 | 1451 AA | | 0.0 |
| | | 0.8264 | 31263 | 1791 BB | | 0.0 |
| | | 0.4958 | 18758 | 1594 FA | | 0.0 |
| | 324 | 0.4958 | 19356 | 1642 AA | | 0.0 |
| | | 7.4377 | 290333 | 1601 BB | | 0.0 |
| | | 4.4626 | 174200 | 1702 FA | | 0.0 |
| E. 0.90 VH | 314 | 0.0160 | 604 | 1725 AA | | 0.0 |
| | | 0.2394 | 9057 | 2108 BB | | 0.0 |
| | | 0.1436 | 5434 | 1810 FA | | 0.0 |
| | 324 | 0.1436 | 5607 | 1922 AA | | 0.0 |
| | | 2.1546 | 84106 | 1815 BB | | 0.0 |
| | | 1.2928 | 50463 | 1962 FA | | 0.0 |
| F. VH | 314 | 0.0138 | 524 | 2181 AA | | 0.0 |
| | | 0.2076 | 7854 | 2356 BB | | 0.0 |
| | | 0.1246 | 4712 | 2066 | | 0.0 |

TABLE XIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| 324 | 0.1246 | 4862 | 2247 AA | | 0.0 |
| | 1.8684 | 72934 | 2036 BB | | 0.0 |
| | 1.1210 | 43760 | 2189 FA | | 0.0 |
| IV.NON-FIRING MANEUVERS | | | | | |
| A.FULL POWER CLIMB | | | | | |
| 1.NORMAL | 0.1000 | 3843 | 1209 AA | | 0.0 |
| | 1.5000 | 57650 | 1576 BB | | 0.0 |
| | 0.9000 | 34590 | 1338 FA | | 0.0 |
| 2.HIGH-SPEED | 0.0017 | 65 | 1786 AA | | 0.0 |
| | 0.0256 | 982 | 1945 CA | | 0.0 |
| | 0.0153 | 589 | 1796 FA | | 0.0 |
| B.MAXIMUM RATE ACCEL | | | | | |
| CLIMB - CRUISE A/S | 0.1870 | 7189 | 2066 AA | | 0.0 |
| | 2.8056 | 107828 | 1949 CA | | 0.0 |
| | 1.6834 | 64697 | 1755 FA | | 0.0 |
| C.NORMAL TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0668 | 2567 | 1050 AA | | 0.0 |
| | 1.0020 | 38510 | 1266 BB | | 0.0 |
| | 0.6012 | 23106 | 1325 FA | | 0.0 |
| (B) 0.7 VH | 0.0668 | 2567 | 1644 AA | | 0.0 |
| | 1.0020 | 38510 | 1929 BB | | 0.0 |
| | 0.6012 | 23106 | 1614 FA | | 0.0 |
| (C) 0.9 VH | 0.0043 | 167 | 1909 AA | | 0.0 |
| | 0.0652 | 2504 | 2051 BB | | 0.0 |
| | 0.0391 | 1503 | 2047 FA | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0668 | 2567 | 1142 AA | | 0.0 |
| | 1.0020 | 38510 | 1350 BB | | 0.0 |
| | 0.6012 | 23106 | 1393 FA | | 0.0 |
| (B) 0.7 VH | 0.0668 | 2567 | 1515 AA | | 0.0 |
| | 1.0020 | 38510 | 1881 BB | | 0.0 |
| | 0.6012 | 23106 | 1771 FA | | 0.0 |
| (C) 0.9 VH | 0.0043 | 167 | 1933 AA | | 0.0 |
| | 0.0652 | 2504 | 2291 BB | | 0.0 |
| | 0.0391 | 1503 | 2211 FA | | 0.0 |
| D. .9 VH CONTR. CORR | | | | | |
| 1.LONGITUDINAL | 0.0033 | 128 | 1950 AA | | 0.0 |
| | 0.0501 | 1925 | 2385 BB | | 0.0 |
| | 0.0301 | 1155 | 1874 FA | | 0.0 |
| 2.LATERAL | 0.0033 | 128 | 1825 AA | | 0.0 |
| | 0.0501 | 1925 | 2002 DA | | 0.0 |
| | 0.0301 | 1155 | 1758 FA | | 0.0 |
| 3.RUDDER | 0.0006 | 22 | 2068 AA | | 0.0 |
| | 0.0085 | 325 | 1784 BB | | 0.0 |

TABLE XIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0051 | 195 | 1904 FA | | 0.0 |
| E.SIDESLIP | 0.0080 | 307 | 1071 AA | | 0.0 |
| | 0.1200 | 4612 | 1115 BB | | 0.0 |
| | 0.0720 | 2767 | 1335 FA | | 0.0 |
| F.PART POWER DESCENT | 0.0040 | 154 | 1923 AA | | 0.0 |
| | 0.0600 | 2306 | 1625 CA | | 0.0 |
| | 0.0360 | 1384 | 1451 FA | | 0.0 |
| V.GUNNERY MANEUVERS | | | | | |
| A.FIRING IN A HOVER | 0.0050 | 192 | 830 AA | | 0.0 |
| | 0.0751 | 2887 | 1062 CA | | 0.0 |
| | 0.0451 | 1732 | 999 FA | | 0.0 |
| B.STRAFING IN ACCEL. FROM A HOVER | 0.0033 | 128 | 1270 AA | | 0.0 |
| | 0.0501 | 1925 | 1194 CA | | 0.0 |
| | 0.0301 | 1155 | 1604 FA | | 0.0 |
| C.GUNNERY RUNS | | | | | |
| 1.PT. TARGET DIVES | | | | | |
| (A) TO 0.6 VL | 0.0187 | 719 | 1129 AA | | 0.0 |
| | 0.2806 | 10783 | 1329 BB | | 0.0 |
| | 0.1683 | 6470 | 1557 FA | | 0.0 |
| (B) TO 0.8 VL | 0.1040 | 3998 | 2003 AA | | 0.0 |
| | 1.5602 | 59963 | 1999 CA | | 0.0 |
| | 0.9361 | 35978 | 2138 FA | | 0.0 |
| (C) TO 0.9 VL | 0.3220 | 12377 | 2498 AA | | 0.0 |
| | 4.8305 | 185650 | 2347 CA | | 0.0 |
| | 2.8983 | 111390 | 2641 FA | | 0.0 |
| (D) TO VL | 0.0008 | 31 | 2991 AA | 94.104 | 0.000000 |
| | 0.0120 | 461 | 2464 CA | | 0.0 |
| | 0.0072 | 277 | 2947 FA | | 0.0 |
| 2.SPRAY FIRE DIVES | | | | | |
| (A) TO 0.6 VL | 0.0080 | 308 | 1244 AA | | 0.0 |
| | 0.1202 | 4621 | 1426 DA | | 0.0 |
| | 0.0721 | 2773 | 1587 FA | | 0.0 |
| (B) TO 0.8 VL | 0.1079 | 4147 | 1968 AA | | 0.0 |
| | 1.6184 | 62202 | 2211 DA | | 0.0 |
| | 0.9711 | 37321 | 2137 FA | | 0.0 |
| (C) TO 0.9 VL | 0.2291 | 8803 | 2710 AA | | 0.0 |
| | 3.4358 | 132050 | 2361 DA | | 0.0 |
| | 2.0615 | 79230 | 2595 FA | | 0.0 |
| (D) TO VL | 0.0040 | 154 | 3291 AA | 24.781 | 0.000006 |
| | 0.0600 | 2306 | 3147 CA | 44.008 | 0.000052 |
| | 0.0360 | 1384 | 3310 FA | 23.132 | 0.000060 |
| D.GUNNERY RUN P/U | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.6 VL | 0.0020 | 77 | 2075 AA | | 0.0 |

TABLE XIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | FREQUENCY OF OCCURRENCE CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0300 | 1153 | 1988 DA | | 0.0 |
| | 0.0180 | 692 | 1878 FA | | 0.0 |
| (B) 0.8 VL | 0.0040 | 154 | 2735 AA | | 0.0 |
| | 0.0600 | 2306 | 2934 BB | | 0.0 |
| | 0.0360 | 1384 | 2726 FA | | 0.0 |
| (C) 0.9 VL | 0.0100 | 384 | 2537 AA | | 0.0 |
| | 0.1500 | 5765 | 3390 CA | 17.566 | 0.000328 |
| | 0.0900 | 3459 | 3137 FA | 45.980 | 0.000075 |
| (D) VL | 0.0067 | 257 | 3522 AA | 11.664 | 0.000022 |
| | 0.1002 | 3851 | 3821 CA | 5.375 | 0.000716 |
| | 0.0601 | 2311 | 3828 FA | 5.289 | 0.000437 |
| 2.TO THE LEFT | | | | | |
| (A) 0.6 VL | 0.0020 | 77 | 1717 AA | | 0.0 |
| | 0.0300 | 1153 | 1983 CA | | 0.0 |
| | 0.0180 | 692 | 2006 FA | | 0.0 |
| (B) 0.8 VL | 0.0040 | 154 | 2567 AA | | 0.0 |
| | 0.0600 | 2306 | 2871 BB | | 0.0 |
| | 0.0360 | 1384 | 3226 FA | 31.724 | 0.000044 |
| (C) 0.9 VL | 0.0100 | 384 | 3562 AA | 10.400 | 0.000037 |
| | 0.1500 | 5765 | 3731 CA | 6.663 | 0.000865 |
| | 0.0900 | 3459 | 3147 FA | 44.008 | 0.000079 |
| (D) VL | 0.0067 | 257 | 3976 AA | 3.826 | 0.000067 |
| | 0.1002 | 3851 | 4083 CA | 3.082 | 0.001249 |
| | 0.0601 | 2311 | 3986 FA | 3.747 | 0.000617 |
| 3.SYMMETRICAL | | | | | |
| (A) 0.6 VL | 0.0002 | 8 | 1950 AA | | 0.0 |
| | 0.0030 | 115 | 2116 CA | | 0.0 |
| | 0.0018 | 69 | 2021 FA | | 0.0 |
| (B) 0.8 VL | 0.0020 | 77 | 2759 AA | | 0.0 |
| | 0.0301 | 1155 | 2824 CA | | 0.0 |
| | 0.0180 | 693 | 2850 FA | | 0.0 |
| (C) 0.9 VL | 0.0033 | 128 | 3654 AA | 8.102 | 0.000016 |
| | 0.0501 | 1925 | 2915 DA | | 0.0 |
| | 0.0301 | 1155 | 3139 FA | 45.577 | 0.000025 |
| (D) VL | 0.0007 | 26 | 3209 AA | 33.948 | 0.000001 |
| | 0.0100 | 385 | 3633 CA | 8.563 | 0.000045 |
| | 0.0060 | 231 | 4141 FA | 2.757 | 0.000084 |
| E.GUNNERY TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.5 VH | 0.0250 | 963 | 1599 AA | | 0.0 |
| | 0.3757 | 14440 | 2170 BB | | 0.0 |
| | 0.2254 | 8664 | 1580 FA | | 0.0 |
| (B) 0.7 VH | 0.0729 | 2803 | 1971 AA | | 0.0 |
| | 1.0942 | 42052 | 2745 BB | | 0.0 |
| | 0.6565 | 25231 | 2821 FA | | 0.0 |
| (C) 0.9 VH | 0.0016 | 61 | 2855 AA | | 0.0 |

TABLE XIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| | 0.0240 | 922 | 2955 DA | | 0.0 |
| | 0.0144 | 553 | 2782 FA | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.5 VH | 0.0250 | 963 | 1410 AA | | 0.0 |
| | 0.3757 | 14440 | 1957 BB | | 0.0 |
| | 0.2254 | 8664 | 1613 FA | | 0.0 |
| (B) 0.7 VH | 0.0729 | 2803 | 1882 AA | | 0.0 |
| | 1.0942 | 42052 | 2371 BB | | 0.0 |
| | 0.6565 | 25231 | 2575 FA | | 0.0 |
| (C) 0.9 VH | 0.0016 | 61 | 2688 AA | | 0.0 |
| | 0.0240 | 922 | 2978 BB | | 0.0 |
| | 0.0144 | 553 | 2866 FA | | 0.0 |
| F. S-TURNS | | | | | |
| 1.AT 0.8 VH | 0.0040 | 154 | 2862 AA | | 0.0 |
| | 0.0600 | 2306 | 2577 DA | | 0.0 |
| | 0.0360 | 1384 | 2899 FA | | 0.0 |
| 2.AT VH | 0.0051 | 197 | 3118 AA | 50.055 | 0.000004 |
| | 0.0769 | 2956 | 2739 CA | | 0.0 |
| | 0.0462 | 1774 | 2386 FA | | 0.0 |
| VI.POWER TRANSITIONS | | | | | |
| A.POWER TO AUTO | | | | | |
| 1. 0.5 VH | 0.0033 | 128 | 1257 AA | | 0.0 |
| | 0.0501 | 1925 | 1095 BB | | 0.0 |
| | 0.0301 | 1155 | 1003 FA | | 0.0 |
| 2. 0.7 VH | 0.0083 | 321 | 1451 AA | | 0.0 |
| | 0.1252 | 4813 | 1398 CA | | 0.0 |
| | 0.0751 | 2888 | 1626 FA | | 0.0 |
| 3. 0.9 VH | 0.0060 | 231 | 2029 AA | | 0.0 |
| | 0.0900 | 3459 | 1837 CA | | 0.0 |
| | 0.0540 | 2075 | 1642 FA | | 0.0 |
| B.AUTO TO POWER | | | | | |
| 1.IN GROUND-EFFECT | 0.0040 | 154 | 1487 AA | | 0.0 |
| | 0.0600 | 2306 | 1469 CA | | 0.0 |
| | 0.0360 | 1384 | 1812 FA | | 0.0 |
| 2. 0.4 VH | 0.0067 | 257 | 1531 AA | | 0.0 |
| | 0.1002 | 3851 | 1465 DA | | 0.0 |
| | 0.0601 | 2311 | 1219 FA | | 0.0 |
| 3. 0.6 VH | 0.0324 | 1246 | 1901 AA | | 0.0 |
| | 0.4862 | 18688 | 1564 DA | | 0.0 |
| | 0.2917 | 11213 | 1994 FA | | 0.0 |
| 4. MAX AUTO A/S | 0.0012 | 46 | 2133 AA | | 0.0 |
| | 0.0180 | 692 | 2270 CA | | 0.0 |
| | 0.0108 | 415 | 1836 FA | | 0.0 |
| VII.AUTOROTATION | | | | | |

TABLE XIII (Continued)

| FLIGHT CONDITION | FREQUENCY OF OCCURRENCE PCT. TIME | CYCLES IN 100 HRS. | OSCILLATORY RESULTANT MOMENT @ STA 2.65 | CYC. TO FAILURE X 10**(-6) | DAMAGE FRACTION |
|---|---|---|---|---|---|
| A.STABILIZED FLIGHT | | | | | |
| 1. 0.4 VH | 0.0043 | 167 | 1181 AA | | 0.0 |
| | 0.0652 | 2504 | 921 BB | | 0.0 |
| | 0.0391 | 1503 | 956 FA | | 0.0 |
| 2. 0.6 VH | 0.0809 | 3111 | 1184 AA | | 0.0 |
| | 1.2142 | 46664 | 1061 CA | | 0.0 |
| | 0.7285 | 27998 | 1518 FA | | 0.0 |
| 3. MAX AUTO A/S | 0.0040 | 154 | 2014 AA | | 0.0 |
| | 0.0600 | 2306 | 1988 DA | | 0.0 |
| | 0.0360 | 1384 | 2126 FA | | 0.0 |
| B.AUTO TURNS | | | | | |
| 1.TO THE RIGHT | | | | | |
| (A) 0.4 VH | 0.0033 | 128 | 1428 AA | | 0.0 |
| | 0.0501 | 1925 | 1076 DA | | 0.0 |
| | 0.0301 | 1155 | 1027 FA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 2080 | 1750 AA | | 0.0 |
| | 0.8119 | 31205 | 1771 DA | | 0.0 |
| | 0.4872 | 18723 | 1350 FA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 77 | 2317 AA | | 0.0 |
| | 0.0300 | 1153 | 1927 CA | | 0.0 |
| | 0.0180 | 692 | 1892 FA | | 0.0 |
| 2.TO THE LEFT | | | | | |
| (A) 0.4 VH | 0.0033 | 128 | 1378 AA | | 0.0 |
| | 0.0501 | 1925 | 1050 DA | | 0.0 |
| | 0.0301 | 1155 | 1006 FA | | 0.0 |
| (B) 0.6 VH | 0.0541 | 2080 | 1817 AA | | 0.0 |
| | 0.8119 | 31205 | 1522 CA | | 0.0 |
| | 0.4872 | 18723 | 1501 FA | | 0.0 |
| (C) MAX AUTO A/S | 0.0020 | 77 | 2167 AA | | 0.0 |
| | 0.0300 | 1153 | 2035 DA | | 0.0 |
| | 0.0180 | 692 | 1991 FA | | 0.0 |
| C.AUTO LANDING | 0.0040 | 154 | 1333 AA | | 0.0 |
| | 0.0600 | 2306 | 1347 DA | | 0.0 |
| | 0.0360 | 1384 | 1494 FA | | 0.0 |

ENDURANCE LIMIT = 2980.0 TOTAL DAMAGE (D) = 0.013179
MATERIAL = ALUM
FREQUENCY = 2 / REV OF M/R FATIGUE LIFE = 100/D = 7587 HOURS